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INFORMATION SERIES

R66FPD56

TITLE:

HYPERSONIC RAMJET RESEARCH ENGINE - COMBUSTOR DESIGN

(NASA-CR-155078) HYPERSONIC RAMJET RESEARCH ENGINE: COMBUSTOR DESIGN (General Electric Co.) 119 p

N77-85502

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TECHNICAL INFORMATION SERIES

Title Page

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SUBJECT CLASSIFICATION
Combustion - Supersonics - Hydrogen

NO. R66FPD56

DATE February,1966

TITLE

Hypersonic Ramjet Research Engine Combustor Design

ABSTRACT A preliminary aero-thermo design of the combustion system for the NASA Hypersonic Ramjet Research Engine has been completed. The preliminary design effort was preceded by a conceptual design study in which possible combustor geometries for alternate engine systems were examined. The work depended heavily on the experimental background accumulated at GE on supersonic & subsonic hydrogen combustion.

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CONCLUSIONS

A preliminary combustor design adequately meets the mission requirements, and the combustor's cycle performance and compatibility with other engine components is substantiated by experimentally verified design criteria.

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For list of contents — drawings, photos, etc. and for distribution see next page (GT 2063-B1).

INFORMATION PREPARED FOR NASA

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GT 2063-A1 (10-52)

TABLE OF CONTENTS

TABLE OF CONTENTS (Continued)

TABLE OF CONTENTS (Continued)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Continued)

HYPERSONIC RAMJET RESEARCH ENGINE COMBUSTOR DESIGN

by

Milton J. Kenworthy, William C. Colley and Philip T. Harsha

SUMMARY

A preliminary aero-thermo design of the combustion system for the NASA Hypersonic Ramjet Research Engine has been completed. This combustion system includes a subsonic hydrogen burner for use at flight Mach numbers from 3 to 6, and a supersonic combustion burner for the flight Mach numbers from 5 to 8. The preliminary design effort was preceded by a conceptual design study in which possible combustor geometries for alternate engine systems were examined. Both the conceptual design and the preliminary design depended heavily on the experimental background accumulated at General Electric on supersonic and subsonic hydrogen combustion.

The preliminary combustor design adequately meets the mission requirements, and the combustor's cycle performance and compatibility with other engine components is substantiated by experimentally verified design criteria.

INTRODUCTION

The objective of the present work on the NASA Hypersonic Ramjet Research Engine is to define the best possible research engine for operation between Mach 3 and 8 which has subsonic and supersonic combustion capability. This effort is defined in detail in the NASA "Statement of Work" for Phase I of the "Hypersonic Ramjet Experiment Project, Conceptual and Preliminary Design of the Hypersonic Ramjet" (Reference 1).

The specific effort on aero-thermo combustion supplied conceptual designs for a variety of engine systems so that the best engine system could be selected. After the engine concept was selected from an overall systems standpoint, the detailed aero-thermo design of the combustor was completed.

The experimental background and design methods for supersonic combustion have been developed only in the last few years. This report begins with an outline of General Electric's experimental background in supersonic combustion. This experimental background is the justification for the design features adopted in the conceptual and preliminary design efforts.

EXPERIMENTAL BACKGROUND FOR DESIGN

Large Rectangular Combustor Tests

During 1962-1963 several supersonic combustors having rectangular cross-sections were tested in the Development Test Sub-operation's Cell 5. These combustors were 3.5 to 5 inches in width and 6 to 13 inches in height. The air supply was heated indirectly to 1460°R, then further heated by burning with hydrogen to temperatures up to 3000°R. Of these tests, the two considered most significant are described below:

Wall injection combustor - The flowpath of this combustor is shown in Figure 1. The heated air was expanded through the large area ratio nozzle shown. At the plane where the air Mach number was about 2.0, a film of hydrogen was introduced on the top and bottom walls. A constant-area rectangular duct 59 inches long followed the diverging section, exhausting to atmosphere.

This test is described in ASD-TDR-63-193 (Reference 2).

Wall static pressure measurements made during operation of this combustor are shown in Figure 2. The measurements with no fuel injected showed that the boundary layers in the constant area duct separated before the flow reached the exit, as evidenced by the rise in static pressure toward atmospheric. The separated regions were sufficient to initiate combustion when fuel was injected, despite the relatively low air stagnation temperature (2500°R). Combustion caused the separation to reposition itself farther upstream than its nonburning location. With sufficient combustion, the separation stabilized in the diverging portion of the duct.

Figures 3 and 4 show profiles of local Mach number and equivalence ratio derived from gas samples and impact pressure measurements taken along the vertical centerline of the duct, 13 inches from the injector. With sufficient combustion to drive the separations upstream of the measuring station, the separated regions appear in the Mach number profiles as large regions of low Mach number adjacent to the walls. The equivalence ratio profiles show that most of the fuel was concentrated in the separated regions.

From this test it was learned that reliable autoignition data in supersonic streams could not be obtained unless the facility exhaust pressure were low enough to assure full flow in the duct prior to combustion. It was also evident that an observed pressure rise in a combustor did not necessarily mark the location of autoignition of the fuel-air mixture, as it could also be caused by combustion-induced boundary separation with recirculation-type flame stabilization.

Strut injection combustor - The "strut" of this combustor spanned its narrow dimension, and extended forward through the throat of the air supply nozzle, as shown in Figure 5. Thus, the strut was effectively without a leading edge. Hydrogen was injected from elongated jets in the sides of the strut near its trailing edge, as shown in Figure 6. Provision was made to heat the hydrogen by burning directly with oxygen.

This test is described in APL-TDR-64-21 (Reference 3).

Wall static pressures measured during operation of this combustor are shown in Figure 7. With room temperature fuel, the pressure rise indicating onset of combustion occurred well aft of the injector, the distance decreasing with increasing fuel flow. Combustion was presumably stabilized by a separated boundary layer. With heated fuel, combustion occurred in the vicinity of the fuel injector, and the steep pressure gradients indicative of separation were absent.

Fuel concentration profiles measured 18 inches aft of the injector (Figure 8) showed appreciable jet penetration. Mach number profiles (Figure 9) measured near the combustor exit showed fully supersonic flow, even with boundary separations present further upstream in the burner.

Cylindrical Combustor Model Tests

During 1964 and 1965, a series of water-cooled models of supersonic combustors were tested in the Hypersonic Arc Tunnel facility, in a program aimed at empirically developing an aerothermodynamic design suitable for application to a ramjet engine for flight speeds between Mach 6 and Mach 12. The models were of circular cross-section, with a three inch inlet diameter. Air from the arc heater was introduced at Mach numbers of 2.75 and 3.25, static pressure normally 7 psia, and stagnation enthalpies simulating various flight speeds from Mach 5.5 to Mach 10.1. Gaseous hydrogen fuel was introduced at room temperature. The combustors exhausted to an evacuated chamber. Combustor measurements consisted principally of wall static pressure and gas samples and impact pressure surveys at the discharge plane.

This program is documented in APL-TR-65-103 (Reference 4).

The combustor models tested had exit areas larger than their inlet areas. The models were classified according to the manner in which the area diverged. The first group, termed step combustors, took the entire area change at a single plane near the burner entrance. The second group, the conical combustors, had walls that diverged in a straight taper over most of the length of the burner. The step-cone combustors combined features of the first two groups.

Step combustors - Typical step combustor configurations are sketched in Figure 10. Wall static pressure measurements made during operation of one such configuration are shown in Figures 11 and 12. It was found that the pressure on the base of the step was normally low, increasing only when sufficient combustion occurred to choke the combustor exit. This was readily accomplished at the lower enthalpy levels of Figure 11, but required higher fuel flows at the higher enthalpy levels of Figure 12. The disturbances to the flow negotiating the step had a beneficial effect on mixing, producing high combustion efficiencies shown in Figure 13. Despite the high efficiency, the overall thrust potential was degraded by the low step base pressures at high enthalpy levels, as shown in Figure 14.

With air enthalpy levels below about 1200 BTU/lbm, autoignition of the room temperature hydrogen did not occur, even though the fuel jets were directed normal to the air stream and a large wake region was provided behind the step. It was found that ignition could be accomplished by elevating the pressure in the chamber to which the combustor exhausts to separate the combustor boundary layers. If sufficient combustion were established to choke the burner exit, the burning would remain stable when the exhaust pressure was again lowered. This technique was suitable for igniting test combustors, but is impractical for flight combustors in engines without variable exhaust nozzles.

Several flightworthy ignition techniques were identified, including a short-duration solid-propellant gas generator. This cartridge injected a charge of hot gas into the combustor sufficient to momentarily choke the flow, initiating combustion with air enthalpies of 750 BTU/lbm. The cartridge arrangement is sketched in Figure 15. Also shown are wall static pressure measurements before and after firing the igniter.

Conical combustors - A typical conical combustor configuration is sketched in Figure 16.

The conical combustors functioned satisfactorily only over a narrow range of simulated flight speeds. At low enthalpy levels, with fuel injected just aft of the start of the taper, the pressure gradients induced by combustion were sufficient to separate the burner inlet stream, as shown in Figure 17. This behavior was interpreted to indicate the need for more rapid area divergence for low enthalpy levels. At slightly higher enthalpy levels, the pressure gradients were drastically reduced, as shown in Figure 18, becoming too gentle to promote rapid mixing. Strong pressure gradients were restored by moving the point of fuel injection forward of the start of the taper, but the tendency to separate the inlet returned. This is also shown in Figure 18. At very high enthalpies, strong pressure gradients could not be induced with only one diameter of constant-area duct between fuel injector and start of taper, as shown by the measurements in Figure 19.

Lack of strong flow disturbances to promote mixing resulted in relatively low combustion efficiencies for the conical combustors, as shown in Figure 13. This was compensated by higher wall pressure forces, so that the thrust potential was comparable to that of the step combustors under similar operating conditions (Figure 14).

Step-cone combustors - The step-cone combustors contained a small step near the burner entrance, followed by a straight taper over the remainder of the burner length. These features represented an attempt to find a compromise between the ability of the conical combustors to maintain high wall pressure forces at high enthalpy levels and the resistance to inlet separation of the step combustors. A typical step-cone combustor is sketched in Figure 20.

Improved resistance to inlet separation was realized, as is evidenced by the comparison in Figure 21 of wall pressure measurements with conical and step-cone combustors under similar conditions.

Although wall pressure forces were higher than those of the step combustors, the pressure gradients desired for promotion of mixing still diminished with increasing enthalpy level, as shown in Figure 22. Partial restoration of the pressure gradients was accomplished by advancing the fuel injection farther forward at the step, as shown in Figure 23. It was concluded that high performance over a wide range of flight speeds would require staging of the fuel injection: injection for forward at high flight speeds would provide strong pressure gradients by placing the combustion in a constant-area duct, and injection farther aft at low speeds would provide the area relief required to avoid inlet separation.

Both the combustion efficiency and the thrust potential of the step-cone combustors were generally high at moderately high enthalpy levels, as shown in Figures 13 and 14.

Two-Dimensional Combustor Tests

In 1965 and early 1966, supersonic combustion tests were conducted in burners having rectangular cross sections, 5 1/2 inches wide with burner inlet heights of one-fourth inch and one-half inch. The two-dimensional flow path of these burners closely simulates the geometry of a sector of an annular burner. Figure 24 indicates the shape and fuel injector location in these sector burners. The purpose of these tests was to demonstrate the similarities between two-dimensional burners and burners having circular cross sections, so that the design criteria developed for the circular cross section burners could be generalized to apply to two-dimensional burners or to annular burners. The experimental data from this investigation is reported in R66FPD57, (Reference 5).

Figure 25 compares the wall static pressure measurements from two tests. One test was with the step cone burner just discussed above, and the other test was with a two-dimensional burner. The similarity in the wall static pressure field is obvious. The relative size of the two combustors is depicted in Figure 25, illustrating that the burner similarities exist over a wide range of sizes. The annular contour in the NASA preliminary design engine was made two-dimensional and compared in Figure 26 with two of the tested burner configurations scaled to the same inlet height. The area ratios in the tested burners are similar to those adopted for the preliminary design.

The testing was next conducted on a burner having a smaller inlet height, 0.27 inches compared with the previous 0.5 inch high burners. This test was conducted primarily to be sure that the testing included burner heights as small as those expected in the preliminary design engine at Mach 8 flight conditions. Figure 27 presents the static pressures for this 0.27 inch high configuration;the similarity with the curve in Figure 25 is apparent.

This testing also established some burner blowout limits. Figure 28 presents the low pressure blowout limits obtained for one configuration. The maximum altitude conditions for the X-15 flight envelope are shown for comparison. The blowout limits of the burner are seen to exceed the envelope limits of the X-15.

The testing with these two-dimensional burners also included tests with a subsonic burner installed downstream, in order to investigate ignition at low flight Mach numbers. Figure 29 shows the flow path for this configuration. Figure 30 shows the wall static pressures before and after ignition. Ignition was accomplished with a cartridge very similar to that used for the step combustor ignition tests, but with less propellant charge, 6.2 grains instead of 10.5 grains. The step combustor tests were at Mach six flight conditions while these two-dimensional burner tests were near Mach 3 flight conditions; the air total temperature was below 500°F.

Fuel Penetration Tests

High temperature jet penetration tests - During 1963, extensive testing was performed in the Hypersonic Arc Tunnel facility to study problems associated with injection of hydrogen from a wall into a high-temperature supersonic air stream. This test program was documented in APL-TDR-64-21 (Reference 3). The fuel jets were built into a flat plate with a sharp wedge leading edge. A Mach 3.5 air stream having a total temperature of about 5000°R and a static pressure of about 2.5 psia flowed across the plate.

A wide variety of jet geometry was studied, including single sonic normal jets, transverse rows of round sonic normal jets, round oblique supersonic jets, and supersonic normal slots aligned with the flow. The oblique supersonic jets produced good penetration, Figure 31,and provided fuel-nozzle thrust, but would not autoignite under the conditions employed. The normal supersonic slots penetrated and burned well, Figure 32, but provided no fuel nozzle thrust.

Low temperature jet penetration tests - During 1964, tests were performed to further explore the penetration of gas jets into a supersonic stream. These tests used a Mach 3.25 wind tunnel with a rectangular test section. Air temperature was only slightly above ambient. The penetrating jets were located in a flat plate model that spanned the test section. Penetration measurements were primarily by Schlieren photographs, although some gas sampling and recovery temperature probing was done. A typical Schlieren photo is shown in Figure 33. The injected gases were air and helium. These tests are reported in APL-TR-65-103 (Reference 4).

Numerous jet shapes were studied, including single round sonic and supersonic holes, and long rows of holes aligned with the air flow. Both normal and oblique jets were utilized. The measured penetrations of single sonic jets were correlated by the equation:

$$\frac{(\text{Penetration Distance})}{(\text{Effective Jet Diameter})},\ \frac{Z}{D_e^*} = 16\left\{\frac{(\rho u^2)_j^*}{(\rho u^2)_\infty},\ \frac{(\text{Jet Momentum})}{(\text{Free Stream Momentum})}\right\}$$

The penetrations of long rows of sonic jets were correlated by the equation:

$$\frac{Z}{D_e^*} = 41\ \frac{(\rho u^2)_j^*}{(\rho u^2)_\infty}.$$

While the free stream Mach number was not varied in these tests, the first equation correlated well the data of other investigators who had used a wide range of Mach numbers, as shown in Figure 34.

Other important observations from these tests were:

1. High injection pressures are not necessary to obtain good penetration; large, low-pressure jets penetrate as far as small, high-pressure jets having the same flow.

2. Oblique sonic jets with moderately excessive injection pressure penetrate as well as normal jets.

3. Fully expanded supersonic jets penetrate farther than under-expanded sonic jets.

4. Penetration increases with the fourth root of the injectant temperature.

Subsonic penetration tests - Figures 35 and 36 show the jet penetration correlations determined from choked-jet tests with hydrogen and argon. This work is documented in R64FPD341, (Reference 6). For these tests, the variable y in the figure is the distance to the peak of the full concentration measured normal to the main air flow. These data verify the form of the jet penetration equation presented by Shandarov, (Reference 7) and are used to predict the location of the fuel jet peak at points downstream from the fuel injector.

Mixing Tests

Low-speed mixing tests - During 1963, experimental data were acquired on the mixing of parallel flows of different density gases, with an initial boundary layer in one stream. These tests was documented in APL-TDR-64-21 (Reference 3). The apparatus consisted of an 18 inch, square, low-speed wind tunnel, with parallel secondary flow introduced through a one-inch high step-slot formed in the tunnel floor. Injected gases were air, helium, and Freon-12.

Typical measurements of the width of the mixing zone are shown in Figure 37, for air injection, and in Figure 38, for helium and Freon injection. Shown for comparison is the theoretical curve for constant density. It was concluded that the effect of stream density on the spatial extent of the mixing region was secondary, but that the approach boundary layer had a pronounced effect. The approach boundary layer caused faster-than-predicted mixing with injectant velocity equal to or greater than free-stream velocity. This helped explain much of the difficulties observed in the literature in comparing mixing data in cocurrent flow with simple theories based on the shear between the two streams.

Correlation of supersonic mixing results - The combustion efficiencies from the step combustor mentioned previously were primarily mixing limited. These efficiencies were compared with mixing levels predicted from a shear mixing theory and were found to be in reasonable agreement. Figure 39 shows the data points compared with the theory. This mixing theory accounts for the density and static pressure gradients created by the combustion process. The theory predicts increased mixing for conditions that involve static pressure rise through the burner. This was consistent with the very low mixing rates found with burner tests having low pressure rise or having a static pressure drop through the burner. The explanation of this mixing theory and the derivation of some simplified expressions is documented in APL-TR-65-103 (Reference 4).

Subsonic combustor analysis - Under Contract AF 33(615)-1304, a detailed approach for designing subsonic hydrogen combustors was developed and supported with experimental studies. This work included accumulation of combustor performance data with a variety of fuel injector configurations involving systematic variation of pertinent variables such as fuel injector strut spacing, fuel injector hole size, and burner length. Data were obtained and correlated on the penetration of hydrogen into subsonic streams, and appropriate mixing equations were introduced into a machine program.

Figure 40 shows, in schematic form, the overall plan for the combustor analysis in terms of the geometric and gas dynamic variables. As shown in Figure 40, the elements are connected to yield a combustor performance analysis for a particular set of geometric and gas dynamic variables, the geometric and gas dynamic variables can be used to generate sets of parametric curves.

The combustion model is treated as a step-wise, one-dimensional mixing process. At a specified length, the stream velocity and static pressure are assumed constant (across the burner cross section). The local fuel concentrations are not constant. Each local point in the combustor is assumed to be at thermo-chemical equilibrium with the dissociated gas thermodynamic properties associated with the static pressure, static temperature and fuel-air ratio at the point.

Deleting many of the details, the program uses the following relations:

Mass Balance
Energy Balance (Continuity)
Momentum Balance (for a variable area duct)
Fuel Distribution from Source Geometry
Turbulent Diffusion Mixing
Penetration from Fuel Jet
Spray Bar Pressure Losses
Fuel Injector Orifice Relations
Condensed Thermodynamic Properties of Hydrogen-Air Mixtures
Area Variations

These relations, taken together, provide a means for a numerical iterative calculation of the combustor process.

The fuel from separate jets mix into a uniform mixture by turbulent eddy diffusion. Differentiation of the single point source diffusion equation,

$$f = \frac{W_f \, V}{4\pi \, W_a' \, EX} \; e^{\frac{-R^2 V}{4EX}}$$

Where: f = local fuel-air ratio

R = distance from peak to point under consideration

X = mixing length

V = stream velocity

E = eddy diffusivity

W_{f_1} = fuel flow rate per source

W_a = air flow rate per unit area

Leads to the parallel relations,

$$\frac{d\,(\ln f)}{dR^2} = \frac{-V}{4EX}$$

and

$$f_R = 0 = \frac{W_f\,V}{4\,\pi\,W_a l\;EX}$$

where values of E can be determined by experiment.

The results of the computer program have been verified by combustor test. The predicted fuel distributions are not quite as good as those by test. However, the local experimental efficiencies are below those assumed in the program, so that the net effect is that the program predicts slightly lower overall performance than is found experimentally, (References 6 and 8).

COMBUSTOR CONCEPTUAL DESIGN STUDIES

Introduction

The SCRAMJET component development work performed over the four years preceding the NASA engine study, outlined in the previous section of this report, had resulted in the identification of design criteria for workable supersonic combustor designs. These criteria included the establishment of the concept of a combustor incorporating a wall step, followed by a diverging section, to get optimum area for burning without unstarting the inlet flow; utilization of the concept of fuel injection from a wall and normal to the airstream; utilization of a cartridge ignition device; and the establishment of design criteria involving fuel injector penetration and mixing lengths. Good performance was experimentally demonstrated for combustors which included these design features, as is documented in Reference 4.

The conceptual design study portion of the work performed during this contract involved consideration of all reasonably attractive possibilities for combustor geometries. In evaluating these combustor geometries, one of the primary considerations was achieving the required performance at Mach 8. The performance goal set was high, and achieving the goal was difficult in all geometries considered. From the design criteria evolved through previous systems analyses, it was clear that the smallest permissible combustor flow area would have to be used in the supersonic combustor of the NASA engine at Mach 8. At the same time, the performance goals at Mach 4 led to the requirement that the engine capture a large percentage of the airflow and operate at stoichiometric fuel-air ratios at this flight condition, which meant that at Mach 4 the combustor had to have greater flow passage area than was desirable at Mach 8. Thus the system analyses and criteria developed during component testing required that variable geometry be incorporated into the combustor.

The incorporation of a variable geometry combustor would not, by itself, solve the problem of meeting the performance goals. For example, since wall friction loss results in significant performance loss to the engine cycle, minimizing the surface area and length of the combustor passages also became a governing factor in the combustor conceptual design. Furthermore, the interaction of the combustor with other engine systems had to be considered. Since the engine is regeneratively cooled, a high heat transfer rate leads directly to a high fuel flow rate in order to keep combustor wall temperatures within the limits of available materials. Such a high fuel flow rate either prevents the engine from being tested at low equivalence ratios or requires that some of the cooling flow bypass the engine, thus incurring a performance loss due to heat loss from the engine cycle and also resulting in shortened flight times due to higher fuel consumption. As with friction losses, the solution to the problem of reducing the heat transfer in the engine lies in the direction of reducing the wetted surface area of the combustor. The combustor conceptual design then not only had to satisfy the performance criteria developed by systems analyses and previous component testing, but also had to identify means of reducing the combustor surface area and thus means of minimizing the heat load and friction losses.

The engine concept originally suggested in the Phase I proposal, which will be discussed in more detail below, included both a variable inlet and a variable exhaust nozzle. Very early in the conceptual design study it became apparent that the elimination of this nozzle could result in a significant reduction in cooling requirements, accompanied by acceptable changes in performance. The most significant performance decrements that resulted from the elimination of the variable exhaust nozzle occurred in the subsonic mode, where the lack of a variable nozzle to decrease the flow area at high flight speeds resulted in supercritical operation of the engine inlet and resulting lower performance.

Another function of this variable nozzle was to insure ignition at $M = 3$ flight conditions by creating a high static pressure and subsonic flow in the burner region. The burner could then be ignited by a flame source. With the elimination of this nozzle, satisfactory ignition by a simple flame source could no longer be expected. However, previous component testing had identified an explosive ignition device that was capable of igniting a flowing supersonic stream, and thus a feature of all subsequent combustor concepts was the provision for an ignition source of this type.

When the conceptual design portion of this study was initiated, the supersonic combustors developed and tested at General Electric all had burner inlets of circular cross section, 3 inches in diameter. Thus, some effort was directed toward achieving burners closely related to the circular burners tested during 1964-65, i.e., having burner flow passage heights comparable with the 3-inch diameter burners, and possibly having circular or nearly circular cross section. This would have provided a design having the highest confidence level based on available experimental confirmation. However, early consideration of annular combustor concepts led to the initiation of a component test program using a two-dimensional combustor design to simulate a sector of an annular passage. After these two-dimensional burners with smaller burner passage heights had been successfully tested, efforts to maximize the burner height dimensions were primarily related to high altitude flight limitations and residence time for recombination.

The following discussion of the conceptual design effort for combustors in the NASA engine can be conveniently broken down into the major types studied:

1. Axisymmetric engines such as the original baseline engine.

2. Two-dimensional symmetric designs.

3. Other concepts, both two-dimensional and three-dimensional.

The combustor concepts discussed in the following sections do not necessarily correspond to the various overall engine concepts that were considered during the conceptual design phase of this study. The combustor conceptual design study was intended to investigate how well the criteria that had been developed to meet the performance goals could be applied to many different combustor configurations; thus configurations were considered that do not necessarily parallel those that were extensively studied from an overall engine system standpoint.

Application of Previous Experience to the Combustor Conceptual Design

As mentioned in the introduction to this section, the four years of SCRAMJET component development work that preceded the initiation of the design studies reported here had resulted in the identification of criteria for the design of a high performance supersonic combustor. The discussion that follows will point out those design features developed during component testing that were applied to every combustor concept studied.

All combustor concepts considered featured fuel injection from the combustor wall in the supersonic mode. The feasibility of fuel injection from a combustor wall was conclusively demonstrated during component testing, as is documented in Reference 4. The very real advantage of the demonstration of the feasibility of wall injection and its incorporation in the NASA engine design is that it obviates the necessity for fuel injection struts in the supersonic combustor with their attendant problems of cooling, structural design, and the built-in performance loss involved in the incorporation of a strut in a supersonic stream. Because of the small supersonic burner passage height available in most of the concepts examined, useful fuel injection struts would be too tiny to be structurally possible. With all fuel injected from wall injectors the criteria for attaining adequate mixing can be expressed in terms of duct diameters. The various design concepts were therefore compared using equal burner lengths in terms of duct diameters (length divided by hydraulic diameter at the burner inlet).

The wall step that is included in every supersonic design concept discussed below has been shown, as is discussed in Reference 4, to have important functions. The pressure rise in the combustor that results from supersonic combustion can cause a combustion induced inlet unstart if no provision is made for keeping this pressure rise from propagating upstream. This problem is particularly acute if the supersonic combustor is located close to the inlet throat, as it must be to minimize heat transfer and friction effects. Previous component testing has shown, however that the problem of combustion induced inlet unstarts can be avoided by the incorporation of a wall step near the fuel injector location in the supersonic combustor (see Figure 21).

Component testing and systems analyses have also shown that a diverging-area, supersonic combustor is desirable at low flight Mach numbers and that this divergence should decrease as the flight Mach number increases. The incorporation of variable combustor geometry in a combustor concept, in combination with proper wall contouring, allows this requirement to be satisfied. Finally, autoignition criteria developed during the course of previous testing, and the development of an ignition device, provided the basis for a solution to the ignition problem at the start of conceptual design.

Axisymmetric Designs

Elimination of the exhaust nozzle variability; ignition techniques - The first concept considered, Figure 41, was an axisymmetric design with a variable inlet and nozzle, such as was presented in the Proposal for Phase I. In this concept the supersonic combustor is located in tandem with and upstream of the subsonic burner; both combustors have an annular flow passage. All fuel injectors in this concept are located in the combustor walls, with fuel injection in the supersonic combustion mode being accomplished by injectors located in the outer wall in the region of the step. Additional supersonic combustor fuel injection is accomplished from injectors located in the outer wall of the combustor at the entrance to the constant area section. In subsonic combustion operation, fuel injection is incorporated in both inner and outer walls at the entrance to the annular, constant area burner. The variable exhaust nozzle, incorporated as part of this design, not only optimized performance over the required flight Mach number range, but also aided ignition in the subsonic combustion mode. The nozzle area variation capability acted to provide subsonic flow in the combustion region, thus simplifying subsonic burner ignition.

Early cooling analyses showed that the equivalence ratio required to cool this design was excessively high. This would either prevent the engine from operating at low equivalence ratios or force some of the cooling flow to be bypassed out of the engine, with consequent performance loss due to heat loss from the engine cycle, and reduction in flight time. Calculations showed that a significant reduction in heat transfer could be accomplished by eliminating the variable exhaust nozzle and that the performance decrement that resulted from eliminating this variability was not serious. Since the high heat transfer rate inherent in this concept had resulted in significant problems a fixed nozzle was substituted in the design in this and all succeeding studies.

One of the functions of the variable exhaust nozzle is to aid in ignition of the subsonic combustor at Mach 3 flight conditions. Closing the exhaust nozzle throat raises the static pressure of the airstream and decelerates it to subsonic velocities, which would allow a spark ignition method to be utilized. With a fixed nozzle, alternate methods had to be devised for ignition of the flowing stream in the subsonic combustor at Mach 3. Without the nozzle closure capability, the low pressure of the stream and the high velocity are not suitable for ignition by a simple spark or flame source. However, component testing has demonstrated the suitability of a cartridge-type igniter for igniting a supersonic stream of hydrogen and air, Figures 15 and 30. The cartridge igniter operates by momentarily injecting into the flow of premixed combustion gases a jet of high-temperature, high-pressure gas. When it is fired, the cartridge simultaneously raises the static temperature and static pressure within the combustor and decelerates the gas flow. At the lower velocities and higher pressures produced by the firing of the cartridge, contact with the hot burning gases created by the explosion is sufficient to ignite the H_2-air mixture. A further advantage of this method of ignition is that the combustor can be lit at higher equivalence ratios than were possible with the nozzle closure method of ignition, due to the fact that the combustor exit area during the ignition sequence is larger. With the elimination of the variable nozzle, the cartridge-type igniter was adopted for all subsequent designs.

Barrett arrived at opposite conclusion

Several other approaches were considered to decrease the cooling hydrogen flow requirement for the combustor. In the original design, the combustor passages were annular and essentially axial. In such a passage the surface area is proportional to the radius of the passage from the centerline, and bringing the combustor passage closer to the centerline of the burner would reduce the wetted surface area and hence the cooling requirement equivalence ratio. Bringing the combustor passage close to the centerline has the further advantage of a larger burner height with its associated increased burner stability. These considerations led to the development of the next axisymmetric design, shown in Figure 42.

In the design shown in Figure 42 the supersonic burner passage has dimensions very similar to those of circular burners tested in 1964, giving confidence to the extrapolation of the experimental data to this new configuration. It should be noted that at the time that this combustor concept was being developed, the two-dimensional (annular sector) combustor tests, which successfully demonstrated the applicability of circular burner data to other geometries, had not yet been initiated. In this concept, the supersonic combustor is located entirely in the region of the centerbody support strut, and the hydraulic diameter of each segment of the combustor between struts is the same as that of a 3.3 inch diameter circle, i.e., approximately the same as that of the cylindrical combustors previously tested. This correspondence is shown in Figure 43.

This is crap!

The elimination of the variable nozzle also resulted in a decrease of the length of the centerbody and the incorporation of a circular, 10-inch diameter, subsonic combustor. In this combustor, fuel was injected from the structural struts and centerbody as well as from the outer walls.

Because of the fact that the engine is designed to operate in both the supersonic and subsonic combustion modes, additional fuel injector struts were not used. Since supersonic combustors, which injected all of the fuel from the walls without any injection struts, were successfully developed in 1964-65, fuel injection struts are utilized in the designs considered only when the strut structure is already available for some purpose other than fuel injection.

The upstream burner was designed to have a significantly smaller flow area than the downstream burner, thus providing a cycle advantage at the higher flight Mach numbers. The early portion of the supersonic burner has a variable area as the inlet spike translates, providing the smallest flow area at the highest flight Mach numbers. Performance calculations of the overall engine system showed that a substantial portion of the burning at Mach 8 flight conditions must take place in this small, variable area region, if the Mach 8 performance requirements were to be achieved. This variable area region needed to be a larger fraction of the supersonic burner length.

In order to take advantage of the provision for spike translation to obtain optimum combustor areas as well as produce a further reduction in burner length and thus heat load, the combined burner axisymmetric concept discussed next was studied.

Combined burners - All combustor concepts studied for axisymmetric engines involved translating inlet spikes, which provided the inlet starting capability and optimum inlet contraction ratio characteristics. The translating capability of this spike, combined with contoured flow passages in the combustor region, provide a good opportunity for applying the burner length and area ratio criteria developed during SCRAMJET component studies to the engine design. Figure 44 shows the burner cross-section that resulted from these studies.

Because of inlet design considerations that developed while the initial combustor design concepts were being explored, the inlet throat is considerably further away from the engine centerline than it was in the original concepts. In order to minimize the combustor surface area the combustor flow passage curves toward the centerline much more steeply than it did in the earlier concepts. This greater passage curvature, which has a beneficial effect on the combustor cooling load, also allows a greater combustor height change, and thus area ratio change, to be accomplished within a given spike translation distance than in previously discussed concepts.

In this concept, the subsonic and supersonic combustors are both located in the same region of the engine. Area variations compatible with the conflicting combustor area requirements at Mach 4 and Mach 8 can be accomplished. Passage areas compatible with operation at intermediate Mach numbers can also be achieved by choice of spike translation distance. Furthermore, the combustor walls can be contoured to provide the variation in overall combustor divergence rate that component testing showed was desirable for operation over the Mach number range from Mach 3 to Mach 8.

All fuel injection in this concept is accomplished from the walls, and a smaller number of fuel injection stations is required than with tandem burner arrangements.

Although this concept has the advantage of some reduced heat transfer over others of the axisymmetric concepts studied, it also has disadvantages. For example, the subsonic combustor length in this concept is approximately half that in the tandem burner axisymmetric concepts considered, with associated design risk. The centerbody support strut in the combined burner concept lies downstream of the subsonic combustor, and thus it cannot act as a fuel injector strut or flameholder during subsonic combustor operation. Further, the lack of tandem burners eliminates the recombination zone effect that the subsonic burner in a tandem burner configuration has during supersonic mode operation. Finally, the tandem burner arrangement, in contrast to the combined burner, permits the independent-component development of the subsonic and supersonic burners.

The baseline engine - The final axisymmetric concept to be discussed here, and the concept that was ultimately chosen as the most acceptable from an overall engine system standpoint, is one that combines a great many of the good features of the combustors already discussed, while eliminating some of the undesirable features of earlier concepts.

As can be seen from the cross-section sketch of this concept, Figure 45, the baseline design incorporates tandem burners, with the supersonic burner located in a contoured annular passage between the translating spike and outer shell. The supersonic combustor, incorporating available experience, features: a wall step to prevent combustion-induced inlet unstarts, injection from the passage walls, and a diverging flow passage downstream to the subsonic combustor. In addition, the supersonic combustor passage slopes inward toward the burner centerline, which helps to minimize the wetted surface area of the combustor and maximize the combustor dimensions, while allowing sufficient reaction length for good combustor performance. The translation of the inlet spike also allows variation in combustor dimensions and area ratios compatible with component test experience and system analyses at various flight Mach numbers.

Why?

Heat transfer analyses performed at the time of the development of the combined (subsonic and supersonic) burner concept showed little advantage from a cooling standpoint for combined burners over tandem burners. Further, the tandem burner concept incorporates the additional advantage of providing a recombination zone downstream of the supersonic burner with a resultant performance advantage. Since the concept of combined burners offers very little cooling advantage, but does materially increase combustor design risk, tandem burners were selected for this engine concept.

The subsonic combustor in the baseline design is located downstream of the translating portion of the inlet spike and is thus of fixed geometry. The burner entrance region is annular, with the passage widening as the centerbody narrows, offsetting the area reduction effect of the presence of the centerbody support struts. In this design, the centerbody ends within the subsonic burner, at a point just downstream of the struts, with a consequent reduction in cooling load over the original combustor concept. At this point the subsonic combustor flow passage becomes a cylindrical section, which culminates in a moderate area contraction at the exhaust nozzle throat. Fuel injection holes are located in the side walls of the centerbody strut for low Mach number subsonic burner operation.

A step is located in the outer passage wall at the entrance to the subsonic combustor to act as a flameholder at high subsonic operating conditions. Fuel injection ports are provided in the outer passage wall just upstream of the step, and further injectors are located in the step itself, injecting downstream, and in the centerbody wall opposite to the step, injecting normally.

Ignition in subsonic operation is provided for by the incorporation of cartridge-type igniters just downstream of the struts in the low flight Mach number subsonic combustor. Mode transition capability is incorporated by provision for transition from the fuel injectors located upstream of the step in the subsonic combustor to those located downstream of the step in the supersonic combustor at Mach 5. Ignition of the fuel-air mixture in the supersonic combustor passage will be accomplished through the back-pressuring of the flow in this passage caused by the combustion in the subsonic burner. With the supersonic combustor operating, the subsonic combustor passage will act as a recombination region, thus offsetting some of the performance loss associated with wall friction in this passage.

Although the concept just described is the one that was ultimately selected for the NASA engine design, other configurations also were studied. A description of these concepts is contained in the following sections, in which two-dimensional, symmetric, and other (cylindrical and two-dimensional) asymmetric concepts are discussed. It should be noted that the combustor concepts discussed in this section do not necessarily correlate with various designs that were studied from a systems standpoint. The concepts considered herein were studied in order to provide combustor design information necessary for the final engine design selection, and the final selection was made from an overall system and not primarily a combustor design standpoint.

TWO-DIMENSIONAL COMBUSTORS

In the course of the conceptual design of the NASA Research Ramjet, considerable study effort was devoted to several two-dimensional combustor designs. Conceptually the two-dimensional combustor offers several advantages. Some means of varying inlet and combustor geometry is required but in a two-dimensional design centerbody support struts are not needed in the combustor flowpath; all supporting and translating structure can be built into the side walls. More flexibility may be available in the geometry variations that are possible in the engine and in component experiments. A burner passage height larger than in an annular burner is also of some advantage.

Several possibilities for two-dimensional geometries were considered in this study.

Two-dimensional symmetric burner with hinged walls - As can be seen from Figure 46 this configuration allows large variations in flow area at any given station to be realized. With this configuration the exact area ratios desirable for any given combustor condition can also be obtained. This, plus the fact that the wall mechanisms can be arranged to provide

No Correction

a continuous variation in burner heights, would make this concept especially valuable as a research tool for full scale combustor testing. Additionally, the centerbody in this design can be made to end within the supersonic burner since the area variation capability is provided by hinged walls. The elimination of part of the centerbody length and of struts within the combustor would lead to lower hydrogen flow required for cooling the combustor than in, for example, an axisymmetric burner with a strut supported centerbody. Finally, this design also provides a variable nozzle throat for the subsonic combustion mode, which as was noted in the preceding part of this section cannot be provided in an axisymmetric burner without materially increasing the cooling load.

Two-dimensional symmetric burner with translating spike - This concept, the combustor flowpath cross section for which is shown in Figure 47, sacrifices much of the variable-geometry capability of the hinged-wall concept for the sake of simplicity. Here there is only one moving part, the inlet spike, but a significant amount of area variation capability is available through the proper combination of wall contouring and the translating ability of this spike.

Because of the combined burner feature of this concept, and the shortened spike incorporated to reduce cooling load, it is not possible to obtain a subsonic burner which incorporates an optimum contraction for high Mach number subsonic mode operation. This results in some performance loss, and thus the two-dimensional combustor conceptual design study was directed to identifying a better subsonic burner contraction within the confines of the combined-burner translating spike concept.

Two-dimensional symmetric burner, translating spike nozzle plug - This concept, shown in Figure 48, is basically a refinement of the preceding concept, designed to overcome the objection of non-optimum subsonic burner exit area. Provision for an area contraction at the subsonic burner exit is made by incorporating a contoured plug with the spike. With the spike in the forward (Mach 4) position, the contour of the plug forms a contraction in the annular burner passage. With the spike in position for higher Mach number conditions, the plug extends into the cylindrical section of the burner, removing the contraction from the annular burner section. Thus a performance improvement in subsonic operation is obtained at the expense of an increase in cooling load.

No Correction

Two-dimensional symmetric burner with rotating cowl - This concept, a cross section of which is shown in Figure 49, was studied to assess the effect of reducing the requirements for variable inlet geometry to a very simple, if not crude, form. As can be seen from the sketch, the variation in inlet height and capture area required to meet the performance objectives is accomplished by a simple, alligator-type, rotating cowl. With only this simple variation to permit inlet starting, a combustor configuration was identified that would operate in both subsonic and supersonic modes, through use of staged injection, at the expense of an increase in length over the other two-dimensional concepts previously considered and some performance decrement due to lack of geometry variation in the combustor. This length increase is essentially due to the necessity of placing the subsonic burner in series with the supersonic burner. This

GT 2063-C1 (10-52)

arises from the elimination of the translating spike. Nice conclusion

Asymmetric Designs

In addition to the two-dimensional symmetric and the axisymmetric burner concepts discussed, other combustor concepts with asymmetric layouts were considered. In general, an asymmetric layout has the advantage of allowing larger combustor flowpath dimensions than an equivalent symmetric design, since only one flowpath exists in an asymmetric design (in cross-section) as opposed to two in a symmetric design. From a combustor standpoint, the larger dimensions offer distinct advantages in relating previous experience to the design. This consideration was particularly important before two-dimensional combustor testing was initiated; even after successful two-dimensional test results had been obtained with small burner passage heights, the advantages of larger dimensions with regard to high altitude flight limitations and residence time for recombination remain.

?

<u>Two-dimensional asymmetric combustor</u> - The evolution of the two-dimensional asymmetric combustor shown in figure 50 can be traced directly from the two-dimensional symmetric combustor with translating spike. In the asymmetric design, the translating spike takes the form of an inlet ramp, and the design of a good supersonic combustor is facilitated by the translating ability of this ramp and by the ease of providing the proper wall contour in the combustor. The fact that the subsonic combustor is laid out downstream of the supersonic combustor increases the overall combustor length over the symmetric design originally considered, but the elimination of the center wall in the asymmetric concept partly counterbalances the extra heat load caused by this length increase. The main advantage of this layout lies in the increased passage height dimensions. The mode transition ability desired for a research engine is achieved by staging injection from the downstream injectors to the upstream injectors at the appropriate Mach number, as in the tandem burner, axisymmetric concept.

From the combustor standpoint, this configuration is the best of all the two-dimensional configurations presented here. It is of reasonably simple concept, with a relatively simple area variation mechanism, and the generous combustor dimensions represent the maximum burner reliability.

THREE-DIMENSIONAL DESIGNS

<u>Fixed geometry configurations</u> - The three-dimensional, fixed-geometry designs considered in this study were based on the 6-12 vehicle design study reported in Reference 4. The major conceptual advantage lies in the fact that such an engine could be laid out with a cylindrical centerburner very much like the cylindrical burners that were extensively tested at General Electric in 1964-65. From the combustor standpoint, such a design has the advantage of minimum development effort. However, it was established early in the conceptual design effort that the performance of fixed geometry inlets and combustors could not be optimized to give adequate performance throughout the entire flight Mach number range from Mach 4 to Mach 8.

Variable Geometry Configuration -

The three-dimensional inlet configuration shown in Figure 51 provides an inlet throat closer to the engine centerline than was possible in the axisymmetric configurations. Combustion advantages result from the larger burner passage heights, and a heat transfer and friction loss advantage arises as the burner diverges and the centerbody is greatly reduced in surface area.

CONCEPTUAL DESIGN SUMMARY

During the course of the conceptual design phase of this study, it became apparent that in order to insure good performance there were several design features that should be incorporated in the concept chosen for development in the preliminary design phase. Although the final choice of combustor concept was made from an engine system standpoint, and not primarily from a combustor standpoint, all of the features identified are incorporated in the concept chosen. The conceptual design study further showed that the design features developed during the four years of SCRAMJET component testing at General Electric that preceded this work could be incorporated into every overall engine configuration study, as long as some form of variable combustor geometry was provided. Because the performance goals placed differing design requirements on the combustor at each end of the design Mach number range, if would not have been possible to satisfy the performance requirements without the provision for variable geometry. From a combustor standpoint alone, cylindrical burner tests of 1964-65, showed, as is documented in Reference 4, that for good supersonic combustor performance the combustor passage at low Mach number flight conditions must diverge, to avoid choking the combustor, and that as Mach number increases the combustor passage must approach constant area to retain high performance. Clearly, then, variable combustor geometry must be a feature of the final combustor concept if good combustor performance over the flight Mach number range is to be achieved.

I believe it.

The burner length has been established from component testing at about ten hydraulic diameters. Good combustion efficiency has been measured with these lengths, and the conceptual design studies were made based on this length criteria.

What about shorter ones?

The combustor conceptual design study identified several other features that were important for good combustor performance. For example, the concept of tandem burners was investigated in some detail and found to offer advantages that could not be easily achieved in any other design. Utilizing tandem burners considerably eases the problem of designing good-performance, supersonic and subsonic burners in the same engine. In the combined burner concept, any change that is made in the contours of the supersonic burner to increase performance also affects the subsonic burner; such a change does not necessarily also improve the performance of the subsonic burner. With tandem burners, the subsonic and supersonic burners can be sized separately, consequently greatly reducing the development effort. Further, the use of tandem burners allows combustion mode transition to take place without any change in burner geometry. This is particularly important in a research engine, as the effects of a change in combustion mode can thus be isolated from any change in burner geometry that might otherwise occur simultaneously. An additional advantage is that

Not very much

with the subsonic burner located downstream of the supersonic burner, a recombination zone is provided at the end of the supersonic burner, which materially aids performance.

Although the tandem burner concept requires a larger combustor passage than combined burners increasing friction loss and heat transfer over the latter, system analyses showed that these losses do not outweigh the advantages of a tandem-burner configuration, and thus tandem burners were identified as an important feature to be included in the final combustor choice.

Tied in with the tandem burner concept is the provision for multiple injection points in the combustor. By allowing several points of injection, the combustor is more flexible, since the area at which combustion is taking place at various flight Mach numbers can be controlled by proper injector staging. Multiple injectors can also aid mode transition by allowing a smooth change of fuel from the downstream to the upstream fuel injectors.

Several other features that were identified in the course of component development work undertaken at General Electric were incorporated into every combustor concept considered during the conceptual design study. These features included wall fuel injection, the use of a wall step in the combustor, and the cartridge igniter.

PRELIMINARY DESIGN

A preliminary aero-thermo design of the combustor was completed for the engine concept identified in the conceptual design effort as the "baseline engine".

Figure 52, also available as Drawing Number 4012140-954, is the aerodynamic flow path through the engine. This drawing is the flow path for the structural design drawings, such as Drawing Number 4012140-941 presented in R66FPD59 (Reference 9). Figure 53 is a larger view of the combustion region showing some of the fuel injector details.

Subsonic Burner

Fuel injector, flameholder and igniter locations - The subsonic burner has provision for fuel injection at two axial stations, each of which is intended to operate in a different flight regime. The injectors located at the downstream end of the structural struts operate during the lean operation planned in the flight Mach number range from three to four. The wall injectors upstream of the structural struts operate in the subsonic burning mode through the flight Mach number range from Mach four to six, and are designed to achieve high performance operating at an equivalence ratio of one.

The lean operation planned at Mach three flight conditions is necessary because the burner exit flow area is not large enough to allow combustion to equivalence ratio of one. To obtain good flame spreading and burner stability in an overall lean burner it is desirable to concentrate the fuel in regions that are locally nearer stoichiometric conditions. In a uniform lean mixture the flame propagation velocities are too slow to spread the flame across the burner. The four structural struts provide convenient fuel injectors and flameholders for this lean operation since local fuel concentrations near stoichiometric can be provided in the wake of the struts. Incomplete mixing of fuel and air at the burner exit from these strut fires is not an objection to the design because combustion efficiency is not a design objective in the flight Mach number range from 3 to 4; it is only important that the burner and engine operate in this flight regime.

The four structural struts are too far apart to provide a fuel injection pattern that will achieve a uniform fuel-air mixture at the burner exit. The injectors upstream of the structural struts provide a much more uniform distribution at higher flight speeds where high performance is required at equivalence ratio of one. Fuel is injected from the outside wall and also from the inside centerbody. The annulus region at this fuel injector station is narrower than the spacing between struts providing a better initial distribution of fuel. The additional mixing length for the fuel over that of the strut injection station also results in an improved efficiency. Flame spreading around the circumference is established by a wall flameholder.

The NASA Hypersonic Ramjet Research Engine as presently conceived utilizes these two injector locations only in specific flight ranges. The strut injectors are used at flight conditions below Mach 4 while the injectors upstream of the structural struts are used from Mach 4 to Mach 6. However, if component testing identifies improved injector staging modes, the manifolding, valving, and control system planned for the baseline engine can easily accomodate many kinds of variation. For example, it may be desirable to use both fuel injector stations over part of the flight range; or a superior injector combination for initial ignition of the burner may be found that is different from the optimum combination for steady state operation at the same flight conditions. A very minor modification to the engine control system could permit this possibly desirable feature. A further fuel injector variation that would be investigated in component tests would involve independent control of the fuel from the two walls at the injector station upstream of the structural struts. It is expected that the component tests will show that this last complication will not be needed, and therefore the engine design does not presently contain independent control valves for the two walls.

The wakes behind the structural struts provide a convenient location for igniting the burner. The ignition cartridge which is installed in the outer wall injects a large volume of hot burning gases into the combustible fuel-air mixture in the strut wake. Cross-firing between struts is accomplished through the wake of the centerbody.

Some of the considerations that resulted in the outside wall location for the igniter instead of a centerbody location are:

1. The outside location has easy access for rearming between runs.

2. Multiple engine restarts are provided for by multiple cartridges, one behind each of the four struts. With multiple cartridges in the centerbody, there is increased danger of one charge igniting other charges.

3. Temperature control of the cartridge is more easily achieved at the outside location.

4. Fuel routing through the centerbody is complicated by the presence of an igniter; more space is available on the outside wall.

Design details of fuel injectors, flameholders, and igniters - Fuel injector design details are influenced by considerations of injection pressure, injector spacing, and fuel penetration. Flameholding and spreading characteristics are provided by sizing of the flameholding regions consistent with the conclusions drawn from previous experiments. An ignition cartridge similar to those used successfully in related component tests is incorporated in the design, but the details are not firmly established and require verification by additional component testing.

The fuel injectors were designed to have a critical or supercritical pressure ratio across the injector orifices at all design conditions. This renders fuel flow independent of burner pressure, avoiding undesirable flow fluctuations or maldistribution of the fuel. Since the heat exchanger in the regeneratively cooled walls requires a high fuel pressure level, the back pressure of the fuel injectors does not create a large additional burden on the fuel supply pressure. Switching fuel injector stations with increasing flight Mach numbers permits this fuel injector back pressure to be minimized.

Figure 54 presents the fuel injector pressures along the B-B trajectory, a trajectory just above the low altitude limit of the X-15 airplane. The pressure characteristics are associated with a given total flow area for each injection stage. The injectors for Mach 3 operation consist of 144 .080 dia holes spaced near the trailing edge of the structural struts, while the Mach 4 - 6 injector stage consists of a more complex arrangement having a larger total flow area consistent with the higher fuel temperatures and equivalence ratios at Mach 4 and above.

The 0.25-inch thick, bluff base at the end of the structural strut provides a fuel-injector-flameholding region of comparable size to those tested in subsonic hydrogen burners at General Electric in the past two years. Fuel injection directly into the wake region to improve flame stability has been a feature of many tested designs. Structural methods for including such injectors at the base of the strut have been identified; should component development tests indicate the need for such injectors, they can be readily incorporated.

The flameholding region upstream of the structural struts is provided with wake injection. This flameholder provides flamespreading around the circumference of the burner, and is made 0.25 inch high to be consistent with previous wall flameholder experience. Component tests on subsonic hydrogen burners in 1964 demonstrated the desirability of providing positive flamespreading. In these experiments, a burner with four fuel injector struts was modified by the addition of a one-fourth inch wall flameholder at the intersection of the fuel injector struts with the wall. The previously poor performance, apparently associated with locally unignited regions, was corrected by the crossfiring action of the wall flameholder.

The annular burner height and hence injector spacing at the flameholder station is 1.45 inches. A large collection of performance data are available on subsonic hydrogen burners having injectors that are spaced between one and two inches apart. These data have been correlated and design criteria have been identified and reduced to a machine program, such that the performance of a subsonic hydrogen combustor can be calculated or predicted from a description of the injector configuration. Detailed performance measurements on subsonic hydrogen burners at General Electric have been made only on configurations with fuel injector struts in the freestream, not on configurations where much of the fuel was injected from the walls. Planned component development tests will compare performance achieved from wall injectors with that obtained from freestream injectors to establish the applicability of the extensive data on freestream injectors to wall injection burners.

Most of the fuel from the Mach 4 - 6 injector is injected through holes that are designed to penetrate deep into the flow. To provide a better initial fuel distribution some smaller holes are provided that penetrate a smaller distance into the stream.

As can be seen from Figure 53, the larger holes on the outside wall are made in the shape of slots rather than round holes. This shape was adopted because it fit between the fuel cooling passages more conveniently. Available data indicate that the penetration characteristics of this shape are at least equal to that for the round hole.

The ignition cartridge in the preliminary design for the Hypersonic Ramjet Research Engine is of the same size as a .38 caliber pistol cartridge and contains 10.5 grains of powder. A cartridge of this same size was successfully used to light a supersonic flow in a 4 3/4"-diameter circular duct in 1965 and a smaller charge, 6.2 grains of powder, has since been successfully used to light rectangular or two-dimensional passages at conditions simulating Mach 3 flight. The cartridge is contained in a chamber that is provided with appropriate temperature control. The cartridge in the engine will have an electrical firing mechanism instead of the mechanical firing mechanism used in previous component testing. Development component testing will be necessary to establish the ability of this igniter to function under all of the desired conditions. It may be necessary to use a larger powder charge for the higher air flow conditions which have not yet been tested. Figure 15 illustrates the igniter as installed in a supersonic combustor test in 1965 while Figures 29 and 30 showed the more recent component test where the igniter was used to light a supersonic flow that simulated the supersonic flow through the engine before ignition at Mach 3 flight.

It is expected that at flight conditions above Mach 4, the fire will flash upstream from the struts or from the ignition cartridge. If, however, the fire does not flash upstream in a desirable fashion and if the fire held from the strut flameholders is unsuitable, a simple fix is available. Catalytic igniters consisting of platinum screen similar to that used in the afterburner flameholders of General Electric's J93 engines could be attached to the wall flameholder region. This igniter would be replaced after each flight by a simple bolting technique accomplished from the back of the engine.

Very crude to touching

<u>Mixing and reaction lengths</u> - The burner length is selected to achieve adequate mixing of fuel and air. A 12-inch length from the end of the fuel structural struts provides adequate length for stable burning at Mach 3 - 4 flight, while the 24 inches from the wall flameholder to the end of the burner provides adequate length for high efficiency performance. It has been shown in component tests that this length provides high efficiency for many stoichiometric hydrogen combustor configurations at pressure levels of one atmosphere.

Should depends entirely on fuel injection scheme

The effect of length on the mixing of fuel and air can be estimated from the available machine program calculations or can be observed by direct comparison with the experimental results of similar burners.

Reducing the burner length results in decreased friction loss and heat transfer. Other considerations such as engine weight and engine length are less important. Decreasing the length of the downstream end of the subsonic burner was found to provide only a small reduction in heat transfer and friction loss; therefore, the burner was designed with a conservative length from a combustion standpoint to minimize development effort in achieving a high performance burner.

??? Paper known best?

Chemical kinetic calculations on the final stages of the heat release process indicate that some performance limitations will occur when the burner operates at low pressure. This performance decrement at low pressure, together with decrements accounting for reduced mixing at the lower Reynolds number and pressure levels, are incorporated into the performance estimates used in the cycle calculations, as are the friction loss and heat loss for the particular configuration. The predicted combustion efficiencies for the subsonic mode are presented in Figures 55, 56, and 57. These combustion efficiencies include not only the effects of local incomplete reaction, but also the effects of velocity and temperature profiles, and the increased theoretical dissociation associated with a temperature profile. A minimum in the efficiency curve appears at stoichiometric conditions because nearly complete mixing is required to achieve high efficiency. At either lean or rich conditions completely uniform conditions are not required to achieve an adequate contact between fuel and oxygen.

Preliminary definition of surface contours - The sizing of the fixed geometry exhaust nozzle throat and the associated cross-sectional area of the burner has a very important effect on the overall cycle performance. To obtain high thrust at Mach 4 flight conditions the exhaust nozzle is made large enough to permit stoichiometric burning without causing the engine to spill more than the design spillage flow. If the flow area were made larger than necessary the overall cycle performance would be reduced. Additional total pressure loss would occur in the engine inlet, which would operate supercritical. If the flow area were made smaller, the maximum permissible equivalence ratio would have to be reduced. An exhaust nozzle throat area of 75.12 square inches was selected to permit slightly supercritical operation of the inlet at this flight condition. This area selection was verified with cycle calculations using predicted component efficiencies.

At higher flight speeds optimum engine performance would be obtained with a smaller throat area. Since the throat is fixed, a small performance decrement is suffered at flight speeds above Mach 4. Figure 58 compares the coefficient-of-thrust of the Hypersonic Ramjet Research Engine with the optimum performance that could theoretically be obtained with a variable exhaust nozzle area. The figure shows that the minimum NASA performance requirements are exceeded at Mach 5 flight condition; at Mach 6 flight conditions the performance requirements are met by transition from the subsonic combustion mode to the higher performance supersonic mode. The higher performance of the supersonic combustor at Mach 6 is due to the fact that in the supersonic combustion mode the combustion takes place upstream of the subsonic burner in a flow area smaller than the exhaust nozzle throat, resulting in better cycle performance than for the subsonic mode optimized for Mach 4. Figure 58 also indicates the improved performance that could be theoretically obtained with a variable exhaust nozzle. This performance advantage in the subsonic mode at high flight speeds was sacrificed in the conceptual design study to reduce the engine cooling requirements.

The subsonic combustor itself is sized with a 10% greater flow area than the exhaust nozzle throat. This 10% exhaust nozzle contraction is large enough to provide a stabilizing influence on the subsonic combustor, and is small enough to avoid significant losses in the supersonic flow field that exists in this region of the engine at high flight speeds.

The flow area through the subsonic burner and exhaust nozzle also has an effect on the performance of the engine when it is operating in supersonic combustion mode. The subsonic burner is located downstream of the supersonic burner, and thus with the supersonic burner operating, the subsonic burner becomes a recombination chamber. If a larger flow area were selected for the subsonic burner, the lower static pressure would discourage recombination at high altitudes, while a smaller flow area would result in a higher equilibrium dissociation level at the exhaust nozzle throat.

The transition from annular to circular burner occurs through the structural strut region of the engine. The burner diverges initially by a sudden increase in area at the wall flameholder followed by a more gradual divergence through the strut region. This divergence through the early portion of the strut relieves the form drag on the strut when the supersonic burner is in operation. To further minimize the strut form drag the leading edge is as sharp as is structurally feasible.

In the subsonic combustion mode the friction loss and form drag of the structural struts are accounted for in the cycle calculations by the introduction of a 0.1 velocity head loss. In supersonic mode the friction loss along the burner walls and structural struts is calculated within the cycle programs, and the small form drag or base drag of the structural struts is accounted for by the nozzle thrust coefficient.

SUPERSONIC COMBUSTION

Fuel injector locations - The fuel injectors for the supersonic mode burner are located on the outer wall. Two stages of injectors are provided with separate manifolds and separate control valves. Preliminary component testing in a two-dimensional supersonic burner very similar to the annular burner of the Hypersonic Ramjet Research Engine indicated satisfactory burner operating characteristics with burner inlet air enthalpies from Mach 6 to Mach 8 flight conditions with both of the fuel injector stages operating simultaneously (Reference 5).

In component development testing under Phase II, increased spacing between injector stages will be investigated. This development testing may indicate the desirability of using different injection stages at different flight Mach numbers. At present, the engine control system utilizes both injectors simultaneously, but since separate valving and manifolding is provided in the engine only a minor modification would be required to introduce the increased variability.

Fuel injection from the inside wall was not incorporated in this design; this avoided the difficulties involved in the additional routing of fuel that would be required in the translating spike.

Fuel injector details - The total injector flow area is sized to provide minimum back pressure while achieving sonic injection pressure or greater at all flight conditions. Figure 59 shows the fuel manifold pressures required for the supersonic fuel injectors at the minimum altitude condition for the X-15 vehicle. These pressures, together with the additional pressure drop through the regeneratively cooled walls, are within the capability of the fuel pumping system proposed for this engine.

Figure 53 shows the location of the fuel injector hole pattern consisting of 336, .080 dia holes. The number of holes and the hole pattern was selected to give the optimum fuel penetration. Figure 60 presents the predicted penetration of the fuel jets versus flight Mach number. The calculated penetration of the center of the fuel mass distribution remains near optimum throughout the flight envelope. The decreased penetration associated with high local air Mach numbers at the higher flight speeds is partly compensated by the increased penetration due to higher fuel temperatures and possibly higher equivalence ratios.

Mixing and reaction lengths - The combustor is primarily mixing limited rather than reaction limited. Design efforts emphasize achieving good mixing through good initial fuel distribution, followed by flow areas and resulting static pressure fields that maximize the mixing rates. Chemical kinetic limitations enter into the latter stages of reaction completion and are accounted for by recombination limitations and combustion efficiency decrements at the higher flight altitudes. Other important reaction limits are autoignition limits and blowout limits.

Data from the extensive experimental results reported in Reference 4 on supersonic burners having circular cross sections form the basis for the mixing performance predicted for the Hypersonic Ramjet Research Engine. The predicted mixing performance of the supersonic combustor is based on extrapolation of component test results and is supported by some analytical mixing calculations.

where – fig 39

Figure 39 shows experimental data compared with analytical mixing calculations reported in terms of L/D, the burner length divided by the burner inlet hydraulic diameter. Mixing data from burners of different size should correlate on the basis of L/D. The mixing performance of wall injection combustors that are mixing rather than reaction limited are scalable in size, as long as all dimensions are scaled and the number of fuel injection points is held constant. Reynolds number is the only variable in this transformation, and this variable can be eliminated from the comparison by also transforming the pressure level under consideration. For example, the mixing performance of a 3-inch diameter combustor at 8 psia pressure should be identical to the mixing performance of a 1-inch diameter combustor with one-third of the length at 24 psia pressure. Since free turbulent mixing is to a large extent independent of Reynolds number, this pressure transformation is not very important in the comparison, and very similar mixing would be expected at equal pressures.

This ability to scale supersonic combustors is in contrast to combustor experience in general, where either all dimensions are not scaled, or significant reaction limitations are present. For example it is common when decreasing the diameter of a hydrocarbon-fueled ramjet combustor to decrease the number of fuel injectors and flameholders, rather than to decrease the dimensions of the flameholder and the length of the burner. Reaction limits prevent reducing the flameholder size, and the rigorous scaling criteria for mixing are not met.

In addition to size extrapolation the question of annular versus circular cross section must be considered. Reasonable correlations should be expected when based on hydraulic diameters.

Figure 61 compares a circular burner cross section with a sector of a two-dimensional passage having the same hydraulic diameter. The flow areas pictured are equal, and the fuel injectors penetrate the same distance in the two configurations. The distance between the virtual sources of the fuel are very nearly equal. Since the distance between fuel sources controls the lengths required for mixing, very similar mixing performance would be expected for equal lengths with these two combustor configurations.

The validity of these mixing generalizations is strongly supported by the nearly identical static pressure fields measured experimentally and shown in Figure 26 for two quite different configurations. These two configurations were designed to have similar mixing characteristics.

The extensive test data taken at General Electric on supersonic combustors in 1964 and 1965 (Reference 4) was for burners with an L/D of 10. The burner in the preliminary design varies with flight Mach number but is at an L/D of 10 or greater, justifying the application of the measured performance data from Reference 4 to the preliminary design engine. This measured performance includes not only the effect of incomplete mixing of fuel and air, but also the effects of velocity and temperature profiles. The performance was determined by analytically expanding through a hypothetical exhaust nozzle the individually measured conditions at many points in the burner exit stream. The combustion efficiency of a uniform stream producing the same thrust in this hypothetical nozzle is the efficiency used in the cycle calculations. This method of performance determination was discussed in detail in Reference 4.

Figures 62, 63, 64 and 65 present the combustion efficiencies used in the cycle calculations. Just as with the subsonic burner efficiencies, the minimum in the performance curve near stoichiometric is associated with the need for nearly complete mixing at stoichiometric conditions. The reduced performance at high altitudes is predicted based on chemical kinetic calculations. Conservative rate constants were used in these calculations to account also for the reduced fine-scale mixing that occurs at the lower Reynolds numbers.

Chemical kinetics calculations based on freezing point criteria indicate that the composition is frozen in the exhaust nozzle at all flight conditions. At low altitudes and high flight speeds, significant recombination takes place in the subsonic burner region, resulting in equilibrium conditions at the exhaust nozzle throat. Cycle calculations account for this by assuming recombination to the nozzle inlet at low altitudes. At higher altitudes frozen composition is assumed at the supersonic burner exit without further recombination in the subsonic burner. Combustion efficiency decrements account for the inability to obtain recombination in the latter portions of the ~~burner at the highest altitudes~~.

Doesn't prove worth of cycl. portion

Figure 66 presents a curve representing the thermodynamic equilibrium conditions in stoichiometric hydrogen-air combustion products. The theoretical equilibrium conditions at the burner exit at Mach 8 flight along the B-B trajectory in the preliminary design engine are shown on the curve. The theoretical dissociation level is shown at both the supersonic burner exit and the subsonic burner exit, illustrating the amount of recombination achieved in this region and the heat still unreleased at the exhaust nozzle inlet. The dissociation level is plotted as (unreleased heat of dissociation) divided by the (heating value of the fuel).

The theoretical dissociation level that would exist at the sonic throat of a subsonic burner is shown for comparison with the theoretical dissociation level at the end of the supersonic burner.

Need velocity profile verified

The overall performance of the engine is affected by the rate of heat release as well as by the combustion efficiency. Figure 67 presents the heat release rate predicted from mixing equations that is used in the cycle calculations.

Experimental results from two-dimensional burner tests (Reference 5) indicate that the burner low pressure blowout limits will exceed the altitude limits of the X-15 flight test vehicle. However, autoignition of the burner may be limited to conditions somewhat below the maximum altitude. Further component testing will establish this limit.

Not so.

Chemical kinetic calculations are not very appropriate for predicting autoignition or combustion limits. Ignition and flame stability are associated with local conditions of static temperature and pressure, higher than the one-dimensional free stream conditions. Component tests in 1963 established that combustion could be sustained without flameholders with air and fuel total temperatures below those required for autoignition. The flame propagated by turbulent diffusion from separated recirculation regions next to the walls. Other tests demonstrated that supersonic burners which would not autoignite would remain stable once they were lit. Fuel injection normal to the air stream creates local high-pressure, high temperature regions in which autoignition can occur, thus permitting initial ignition of the stream even though the one-dimensional static temperatures are too low. Once a burner is lit the pressure rise created by the combustion process helps to stabilize the fire by the creation of static temperature levels in the autoignition range.

The predicted ignition delays calculated from experimental data are shown in Figure 68. The effect of pressure rises in the freestream flow indicated to illustrate that autoignition characteristics are attainable over a wide range of Mach numbers in the presence of pressure rises.

It is expected that the burner in the preliminary design engine will autoignite at flight conditions of Mach 6 and above. This is consistent with component test experience. At Mach 5 conditions the fuel will autoignite through the normal shock wave that will exist in the supersonic burner region because of the back pressure created by the subsonic burner.

Figure 69 illustrates the position of a one-dimensional normal shock in the supersonic burner when the subsonic burner is operating. In subsonic combustion mode, a shock recovery process exists in the diffuser section upstream of the burner. This recovery process creates static temperature and pressure conditions suitable for auto-ignition at M = 5 flight. From the position of a hypothetical one-dimensional normal shock in the diffuser, it is seen that this shock recovery process is a significant aid in obtaining autoignition when transitioning to supersonic mode at M = 5 conditions. At Mach 6 flight conditions the shock recovery process is calculated to be near the end of the supersonic burner. However, at Mach 6 conditions autoignition is expected at the fuel injector station.

Burner contour - Highest cycle performance is accomplished by burning in the smallest possible area, thus releasing the heat at the highest possible static pressure and static temperature. Optimization requires different burner flow areas at different flight Mach numbers. At the optimum high temperatures and pressures a significant quantity of dissociated species which prevent the complete conversion of the fuel heating value into sensible temperature exists at the equilibrium conditions. The downstream end of the combustor is made divergent to provide efficient recombination as the static temperature falls in the diverging supersonic flow.

At very high flight speeds, above M = 8.0, the most efficient cycles inject fuel and begin combustion in the minimum throat area provided by the engine inlet system. However, in the flight range from M = 6 to 8, stoichiometric combustion at the inlet throat theoretically must result in a reduction in engine airflow, i.e., increased inlet spillage or an inlet unstart. The supersonic combustor in the engine design, therefore, begins with an initial divergence from the inlet throat.

Figure 70 presents the theoretical equivalence ratio limit that can be burned in the inlet throat area. The curve for the design contraction ratio shows that at Mach 6 flight conditions, burning above an equivalence ratio of 0.3 in the inlet throat area is not permissible. Thus a diverging burner is required. At reduced contraction ratios greater equivalence ratios are permissible. To illustrate that constant area stoichiometric burning at the throat cannot be accomplished with any contraction at Mach 6, a curve is shown for burning in the freestream air without any inlet contraction, a contraction ratio of 1.0. Even at this extreme condition constant-area stoichiometric burning is not possible; the inlet flow will be forced to spill or occupy a larger flow area.

At Mach 8, stoichiometric burning is theoretically just permissible at the design contraction ratio assuming a frictionless combustor. Introduction of friction loss would require a reduction in equivalence ratio.

Achieving the minimum performance required by NASA at Mach 8 flight conditions requires that the burner be optimized for Mach 8 operation. It was not, however, practical to design to this theoretical limit of constant area at the throat. Friction loss, inlet boundary layer separation, and nonuniform fuel or air flow around the circumference require that the Mach 8 burner diverge from the inlet throat area.

Figure 71 shows the minimum burner area ratio at which stoichiometric combustion can take place without affecting the engine inlet flow. The area ratio in the early portion (3 L/D's) of the engine design is also shown on the plot. The engine areas are safely above the theoretical limits.

Further, definition of the early portion of the burner contour was established with the multistream cycle program. The burner contours were adjusted so that the predicted heat release rate at Mach 8 flight conditions did not result in calculated flow Mach numbers as low as sonic anywhere in the burner.

As flight Mach number changes, the inlet spike translates to the position providing the optimum inlet contraction ratio for that flight condition. This same translation is utilized to provide variable geometry through the supersonic burner. The rate at which the area ratio changes with translation distance is affected by the angle toward the engine centerline that is selected for the burner passage. A horizontal burner maintains constant area with spike translation. A large angle toward the centerline results in a decreasing burner discharge area with rearward translation. Thus the large burner area ratios required at M = 5 and M = 6 flight Mach numbers are reduced at M = 8 flight conditions by the rearward inlet spike translation to the Mach 8 design position.

This variable area ratio characteristic for the upstream portion of the burner is shown by the dotted line in Figure 71.

The area ratio of the downstream portion of the burner is designed to provide recombination at Mach 8 flight conditions in a diverging passage as the final mixing of fuel and air progresses. Recombination continues through the subsonic burner region of the engine. This final divergence in the supersonic burner begins after a substantial portion of the combustion has been completed and after a significant static pressure rise has been achieved. Obtaining substantial pressure rise early in the burner is important not only because of its direct effect on the cycle but also because the adverse static pressure gradient increases the mixing rate of the fuel and air.

Figure 72 shows the flow areas through the burner region at Mach 8 conditions.

Combustor Interaction With Inlet and Nozzle

The interaction effects between the combustor and nozzle with respect to profile and recombination are treated in the cycle analysis. The interactions between the inlet and combustor are more complex but are also treated within the cycle insofar as they can be defined. The actual limits for steady state operation and the transient problems must be verified by component testing.

Nozzle Interaction - Cycle calculations treat the recombination through the combustor and nozzle by freezing point assumptions. Calculations not included in the cycle program determine the flow area in the combustor-nozzle region at which the composition effectively freezes; this area is then input to the program.

The combustion efficiency accounts for the effects of major profile effects, since the combustion efficiency was based on experimental data in which the profile was analytically expanded through a hypothetical exhaust nozzle to determine its thrust-producing potential.

The boundary layer along the wall is treated independently of the freestream flow in the multistream program, providing an analytical approach within the cycle program for treating the effect of the combustor boundary layer on the exhaust nozzle performance.

Experiments were conducted in 1965 (Reference 4) on combustor-nozzle combinations; this testing indicated that very little large scale mixing of composition or momentum takes place in the exhaust nozzle. The tests also showed that the composition at the end of the nozzle, measured by gas samples, was consistent with the theoretical equilibrium dissociation level calculated for the conditions at the burner exit. This was also consistent with chemical kinetic calculations which predicted freezing of the composition at the exhaust nozzle inlet.

Interaction with inlet - The cycle calculations, both one-dimensional and multistream, provide indications of excessive engine choking if the combustor under analysis burns too much fuel in too small a flow area to be consistent with inlet flow properties. Calculations not included within the cycle program, such as those indicated in Figure 70 and 71, also provide indications of the inlet-combustor limitations. These theoretical thermodynamic limits are modified by experimental indications of limitations.

Figure 73 shows a comparison of experimental data from inlets and combustors. This figures shows the operating pressure level in the region of the inlet throat step. This pressure must be high enough to assure good burner thrust, but must not exceed burner or inlet stability or choking limits. The lower solid curve on this chart represents the calculated pressure rise ratio from the inlet throat to the combustion region immediately downstream of the throat step as used in the cycle performance calculations. The burner test results from the two-dimensional burner tests (Reference 5) indicate that the cycle values are conservative. The throttled inlet data curve indicates that the burner pressure rise curve is within the pressure rise capability of the inlet.

Angle of attack effects must also be considered in the interaction limits between the combustor and nozzle. The effect is a small perturbation based on the circumferential distribution of flow around the engine inlet throat.

Transient conditions involve interaction problems that have not yet been experimentally studied. These problems occur during burner mode transition, burner ignition, and may be caused by any mechanical vibration or fluid dynamic oscillations which may be present.

Experimental effort is planned early in the Phase II development program to investigate pressure transients during ignition in component tests. These data and steady state combustion data would be compared with inlet capability as in Figure 73. Only after inlets and combustors are tested in combination will it be possible to completely evaluate the interaction problems.

Mode Transition

Transition between subsonic combustion mode and supersonic combustion mode is accomplished simply by opening the valves for the fuel injectors in the supersonic burner.

Initial autoignition of the fuel at Mach 5 flight conditions occurs through the normal shock recovery process that exists in the diffuser region upstream of the supersonic combustion (Figure 69). As the fuel flow gradually increases into the supersonic injectors, the ignition point or flame stabilization point moves upstream towards the fuel injectors. As the fuel to the subsonic burner decreases, the shock recovery process in the diffuser is no longer supported by the subsonic burner back pressure, and the pressure rise process in the supersonic burner is due only to the combustion from the supersonic burner fuel injection.

At Mach 6 conditions the fuel autoignites in the supersonic burners without benefit of the shock recovery process in the diffuser upstream of the subsonic combustion.

The engine control system (Reference 10) provides the means for accomplishing transition at the desired flight condition, and a reverse transition back to subsonic mode can also be accomplished.

REFERENCES

1. Langley Research Center, Langley Station, Hampton, Va., "Statement of Work - Hypersonic Ramjet Experiment Project for Phase I Conceptual and Preliminary Design of the Hypersonic Ramjet", Exhibit A L-4947 April 15, 1965.

2. Flight Propulsion Laboratory Department of the General Electric Company, Evendale, Ohio under Air Force Contract AF33(657)-9361. Scramjet - Mid-Term Report, ASD-TDR-63-193.

3. Advanced Engine and Technology Department of the General Electric Company, Evendale, Ohio under Air Force Contract AF33(657)-9361 "Advanced Air-Breathing Engines, Volume II: Experimental Investigation" Technical Documentary Report No. APL-TDR-64-21, May 7, 1964. Air Force Aero Propulsion Laboratory Research and Technology Division Air Force Systems Command-Wright Patterson Air Force Base Ohio.

4. Aerospace Propulsion Systems Operation, Advanced Engine and Techology Department, General Electric Company, "Analytical and Experimental Evaluation of the Supersonic Combustion Ramjet Engine; Volume II" Air Force Aero Propulsion Laboratory Technical Report No. AFAPL-TR-65-103, December, 1965 (Confidential).

5. D. E. Nieser, Supersonic Combustion Unit, AETD, General Electric Company, "Preliminary Results of Arc Tunnel Testing of a Two-Dimensional Supersonic Combustor" R66FPD57.

6. Advanced Engine and Technology Department, General Electric Company, Cincinnati, Ohio; Advanced Combustion Unit, Phase I Report, "Turbo-Propulsion Combustion System Exploratory Development Program", 1 April 1964 through 30 December 1964 (AF33(615)-1304).

7. G. H. Abramovitch, Moscow, 1960, "Turbulent Jets Theory", (available as Tech. Doc. Liaison Office Unedited Rough Translation, MCL-1256/1 and 2, WPAFB, 1962-ASTIA 28358).

8. AETD, Advanced Combustion Unit, General Electric Company, Cincinnati Ohio, "Turbo-Propulsion Combustion System Exploratory Development Program", Phase II Report, 1 January 1965 through 31 December 1965. R66FPD7.

9. E. E. Halila, N. A. McClean, R. R. Sponseller, G. P. Vaslos, "Hypersonic Ramjet Research Engine Mechanical Design". R66FPD59

10. M. W. French, "Hypersonic Ramjet Research Engine Controls and Accessories Design". R66FPD60

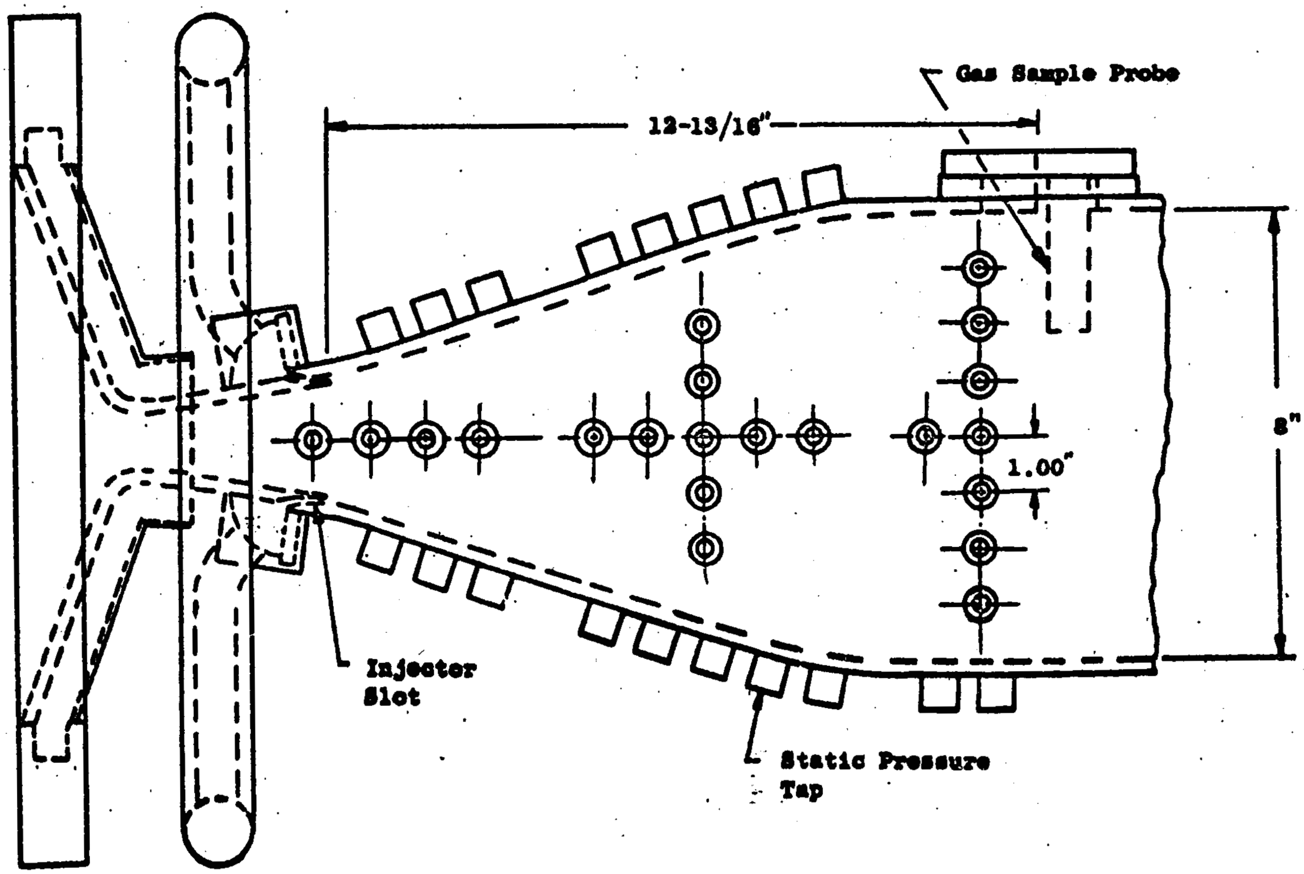


Figure 1 - Schematic - Wall Injection Tunnel

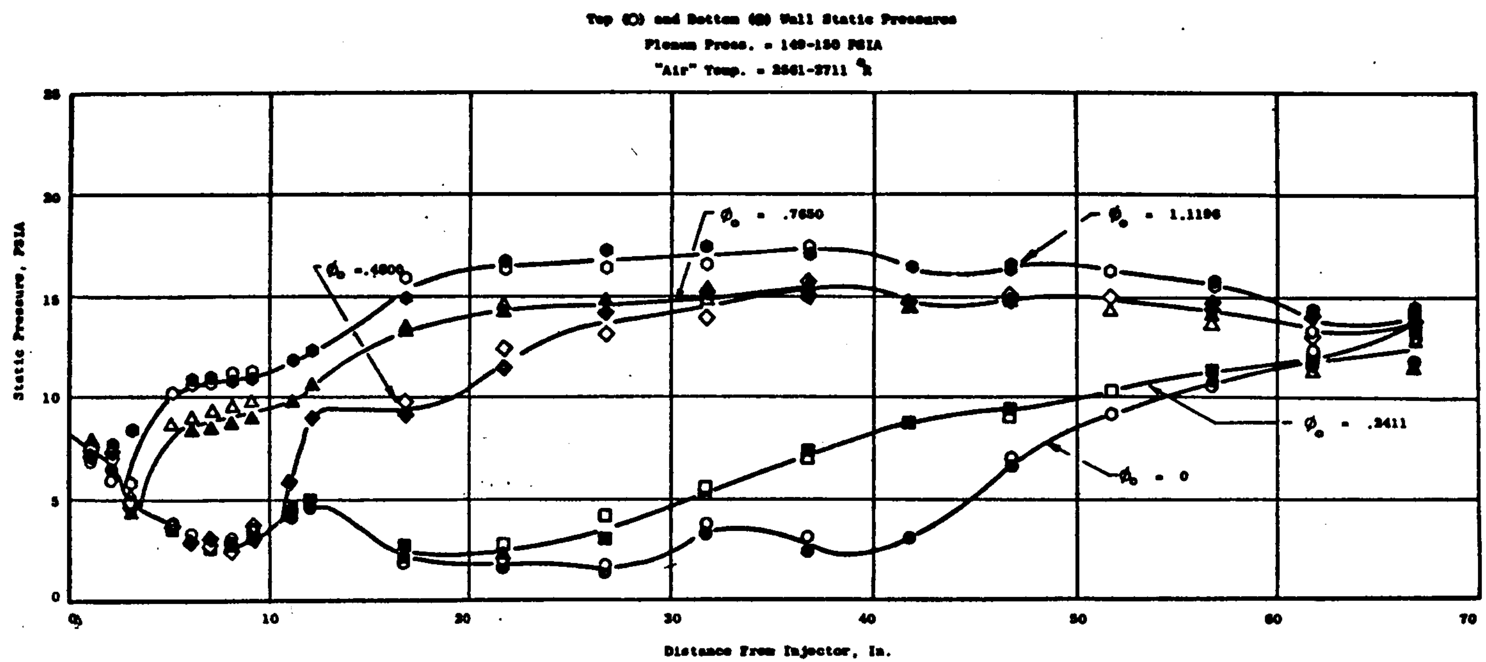
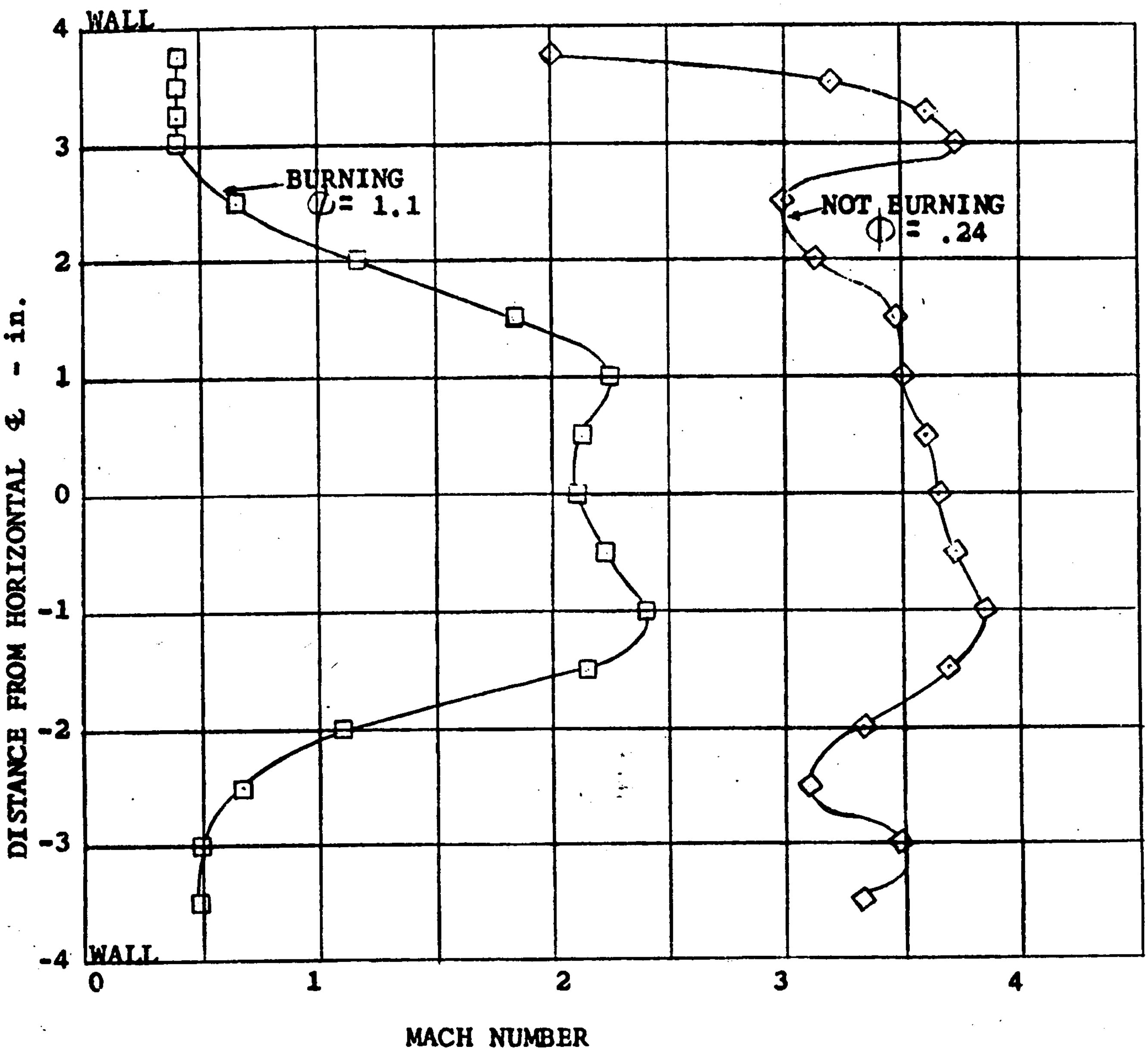


Figure 2 - Wall Static Pressures, Wall Injection Tunnel

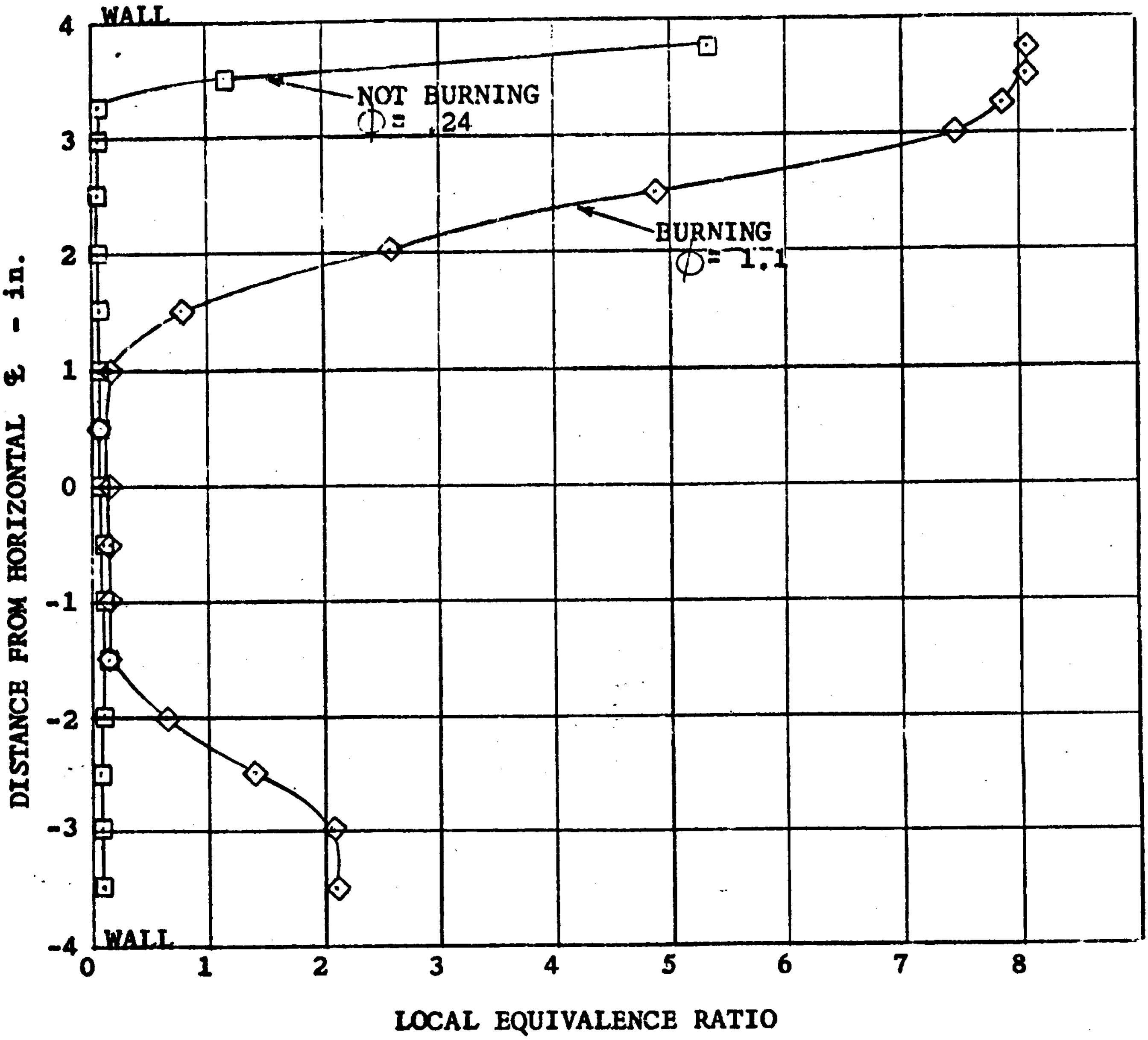
PLENUM PRESS. = 149 150 psia
"AIR" TEMP. = 2660-2730 °R



MACH NUMBER PROFILE FOR WALL INJECTOR

Figure 3

PLENUM PRESS. = 149-150 psia
"AIR" TEMP. = 2660-2730 °R



EQUIVALENCE RATIO PROFILE FOR WALL INJECTOR

Figure 4

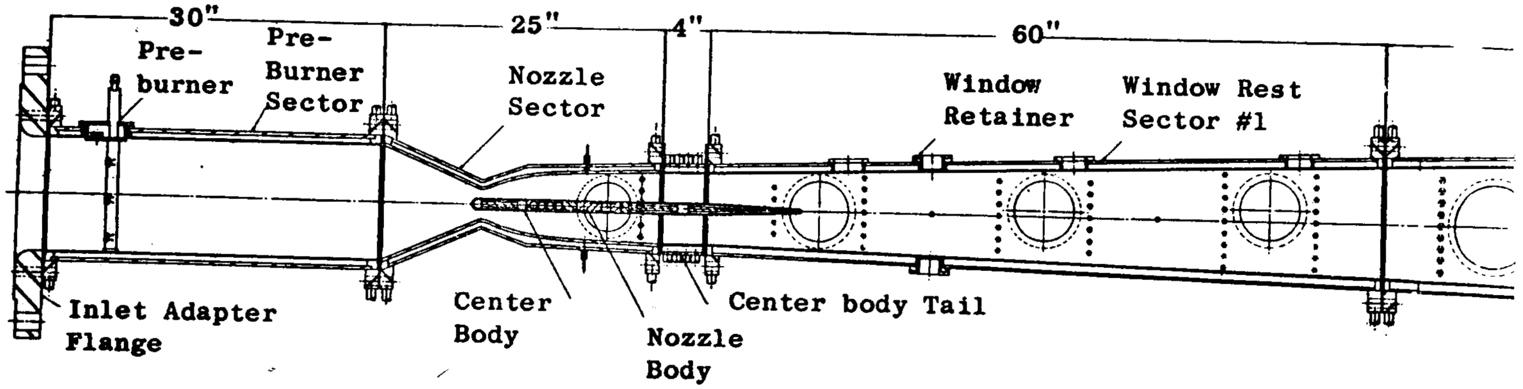
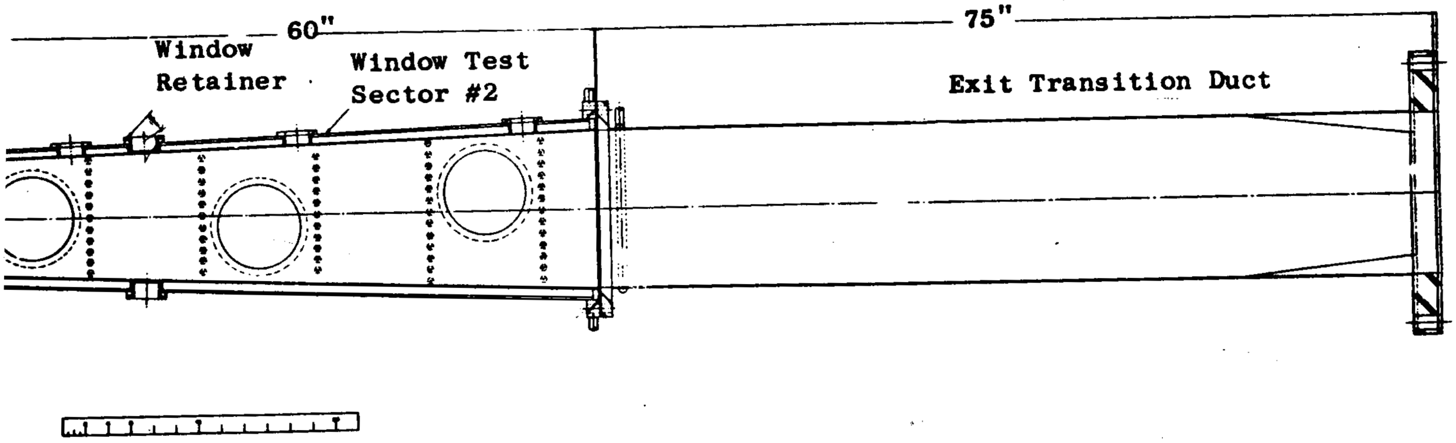


Figure 5 High Pressure,

[redacted]



Scale in Inches

ure, High Enthalpy Tunnel

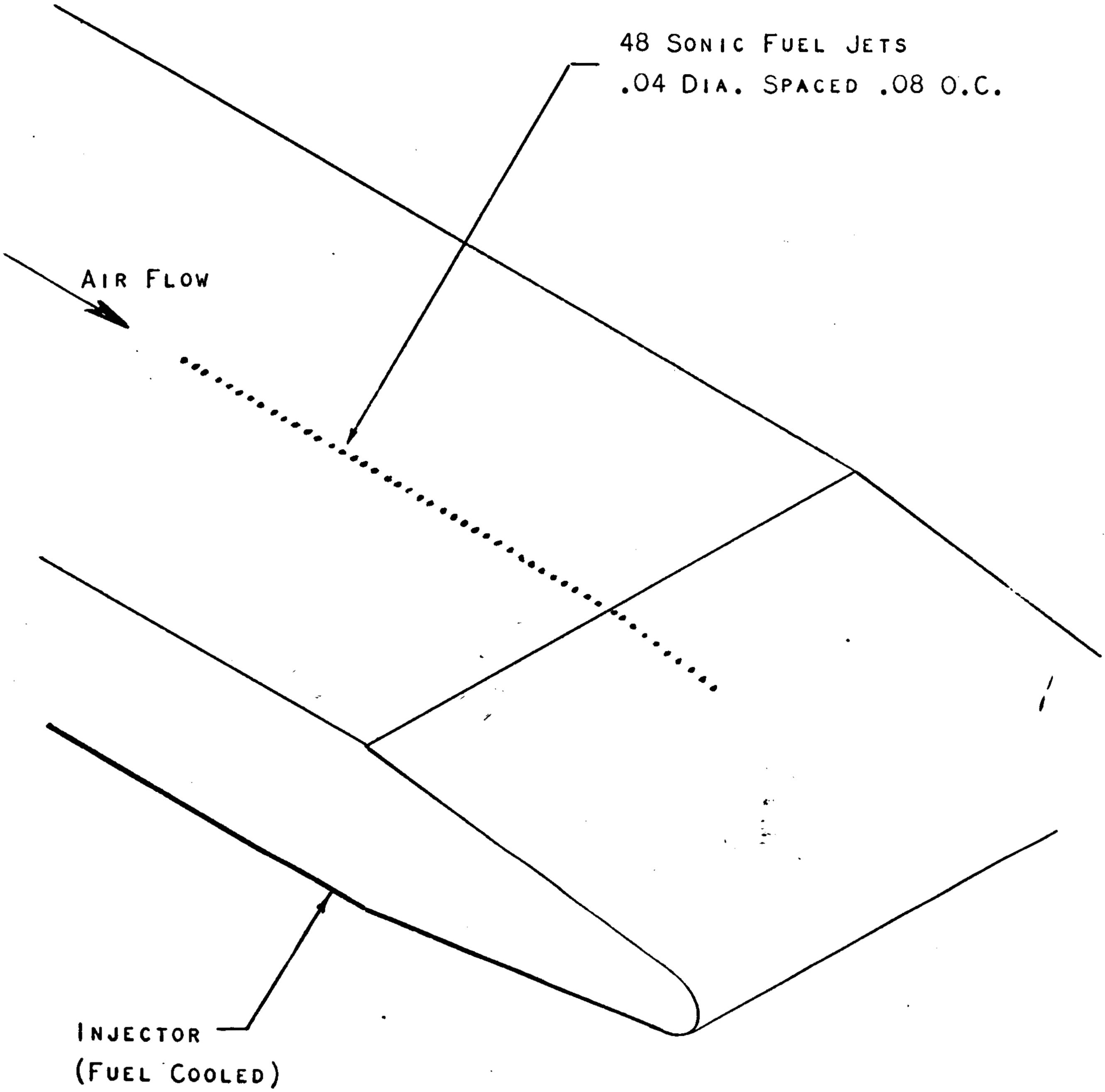


Figure 6: Single Slot Cross-Stream Injector

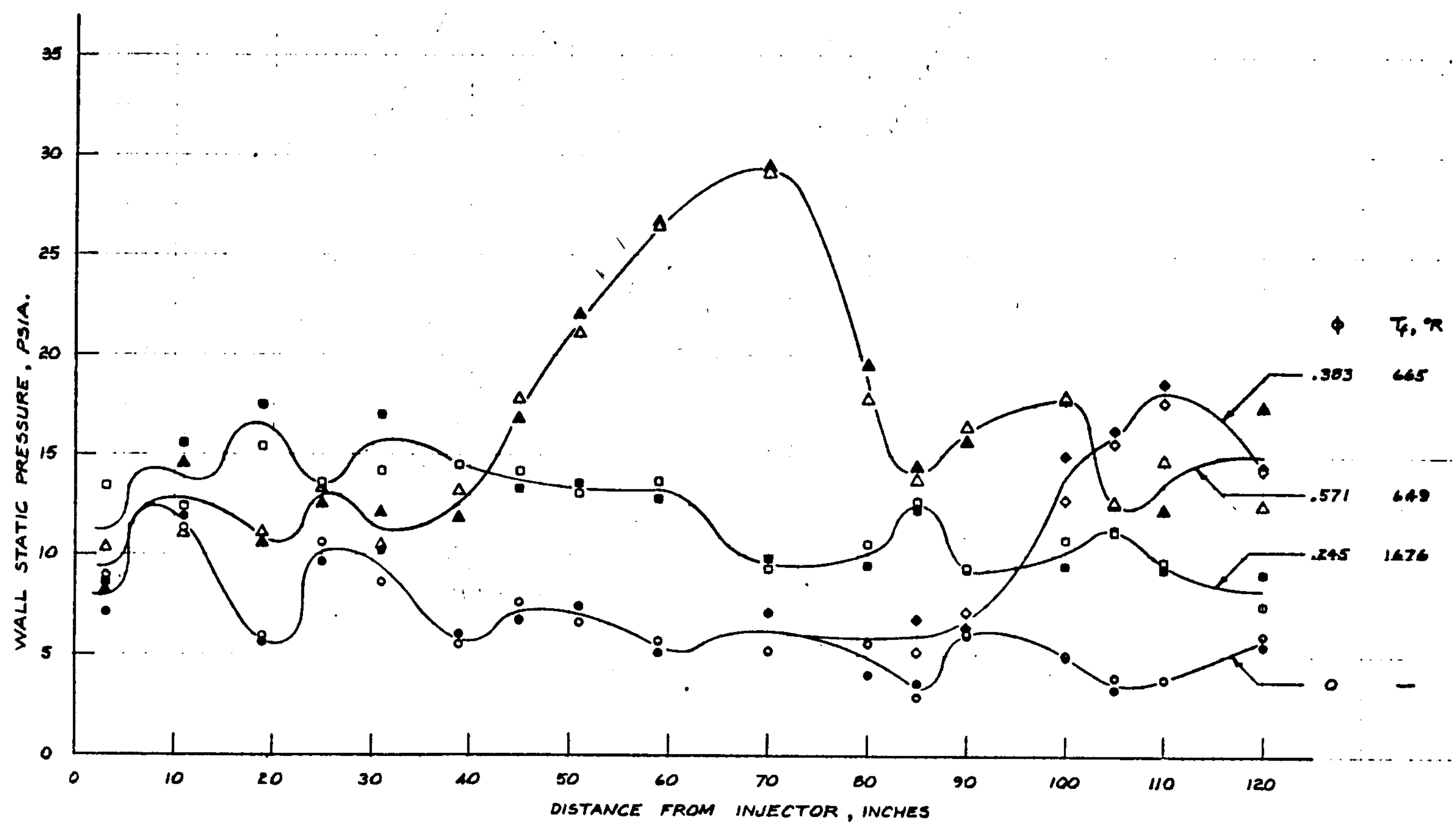


Figure 7: Wall Static Pressures, Single-Slot Cross-Stream Injector

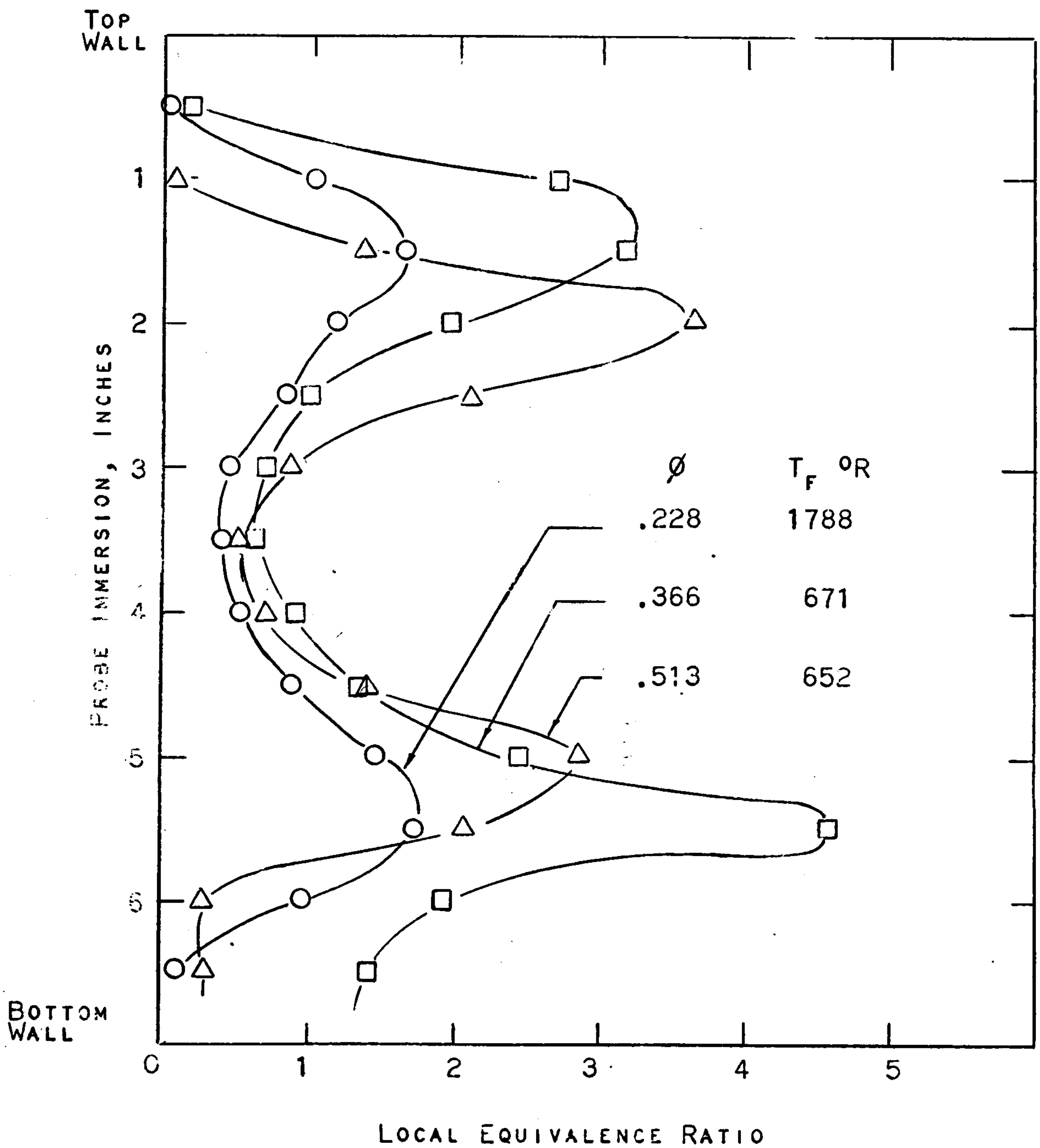


Figure 8: Fuel Profiles 18 Inches from Single Slot Cross-Stream Injector

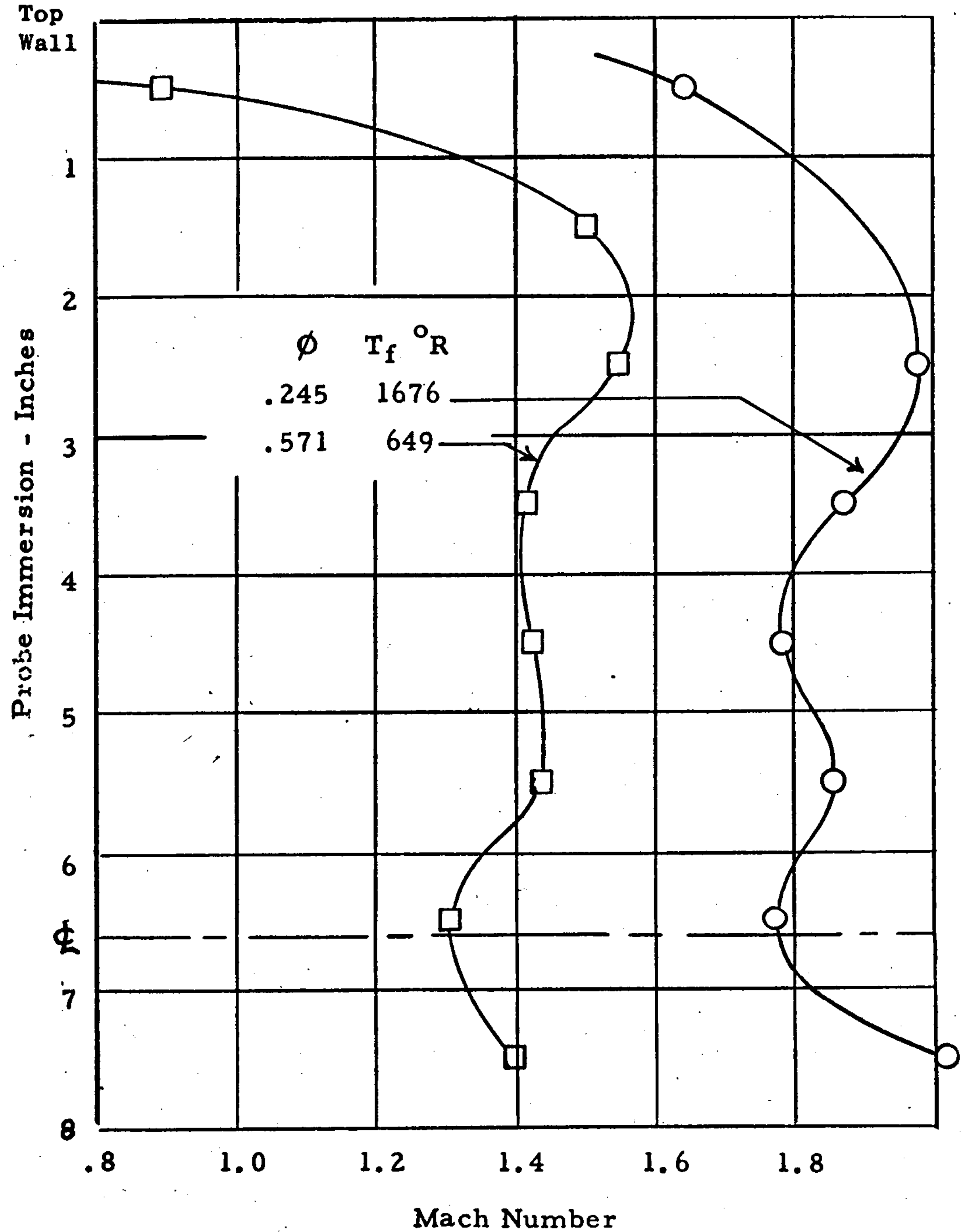


Figure 9. Mach Number Profiles -- 119 Inches from Single Slot Cross-Stream Injector

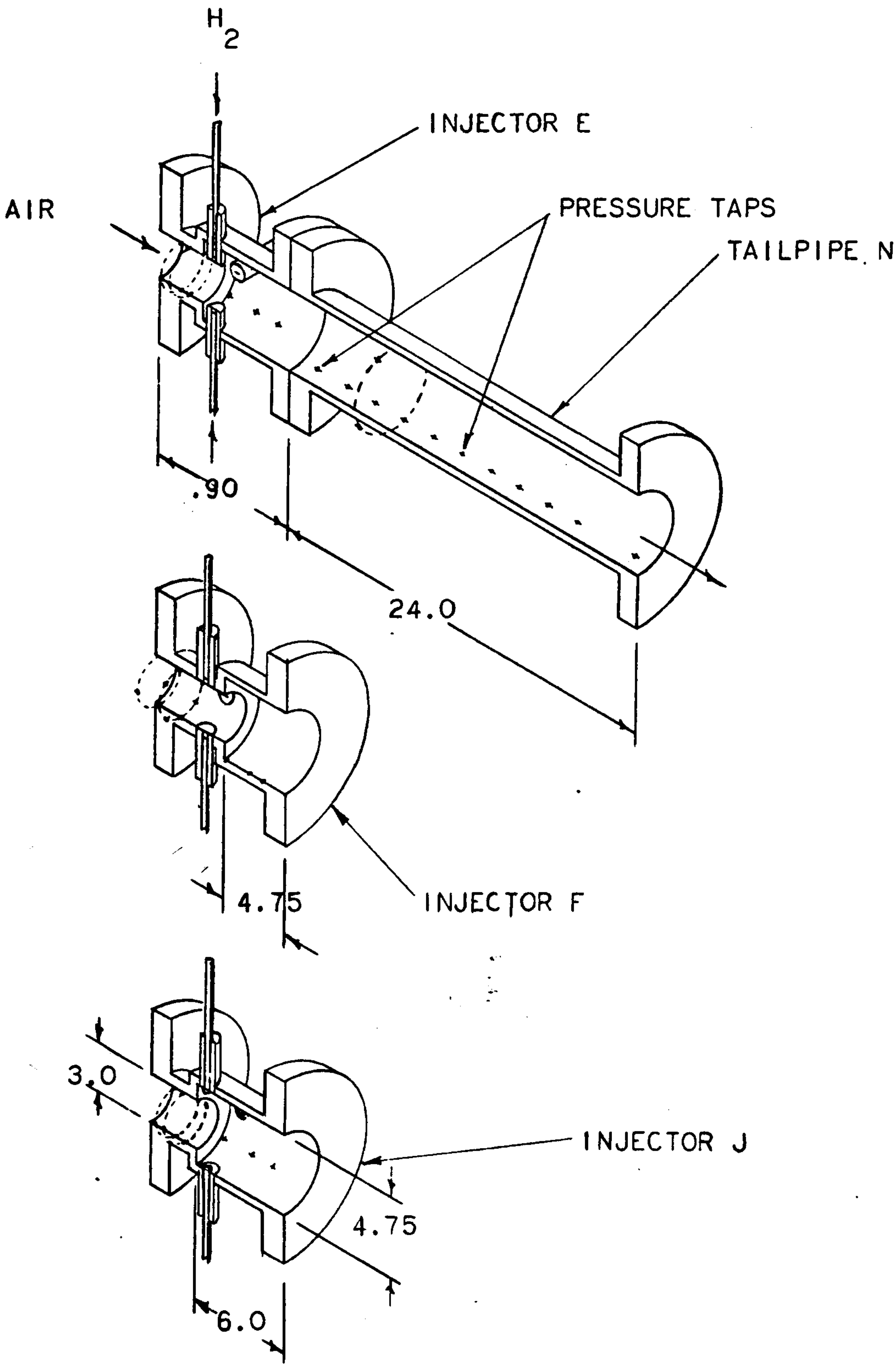
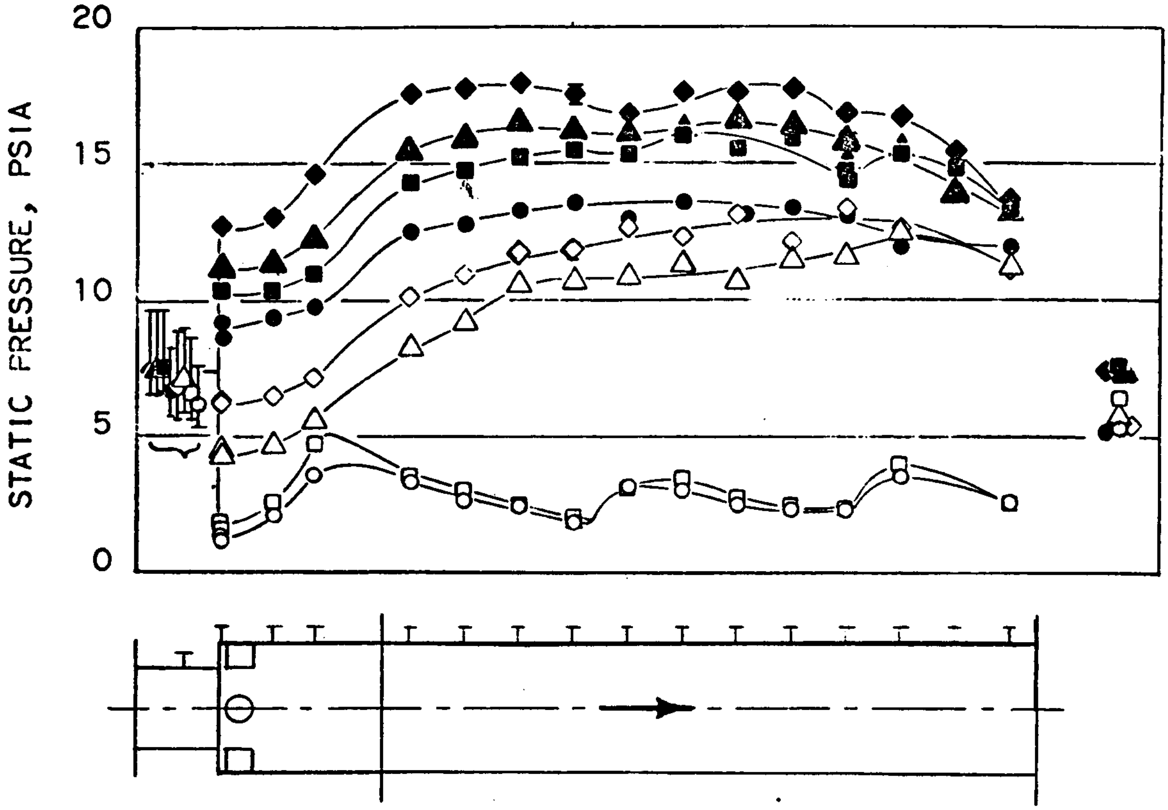
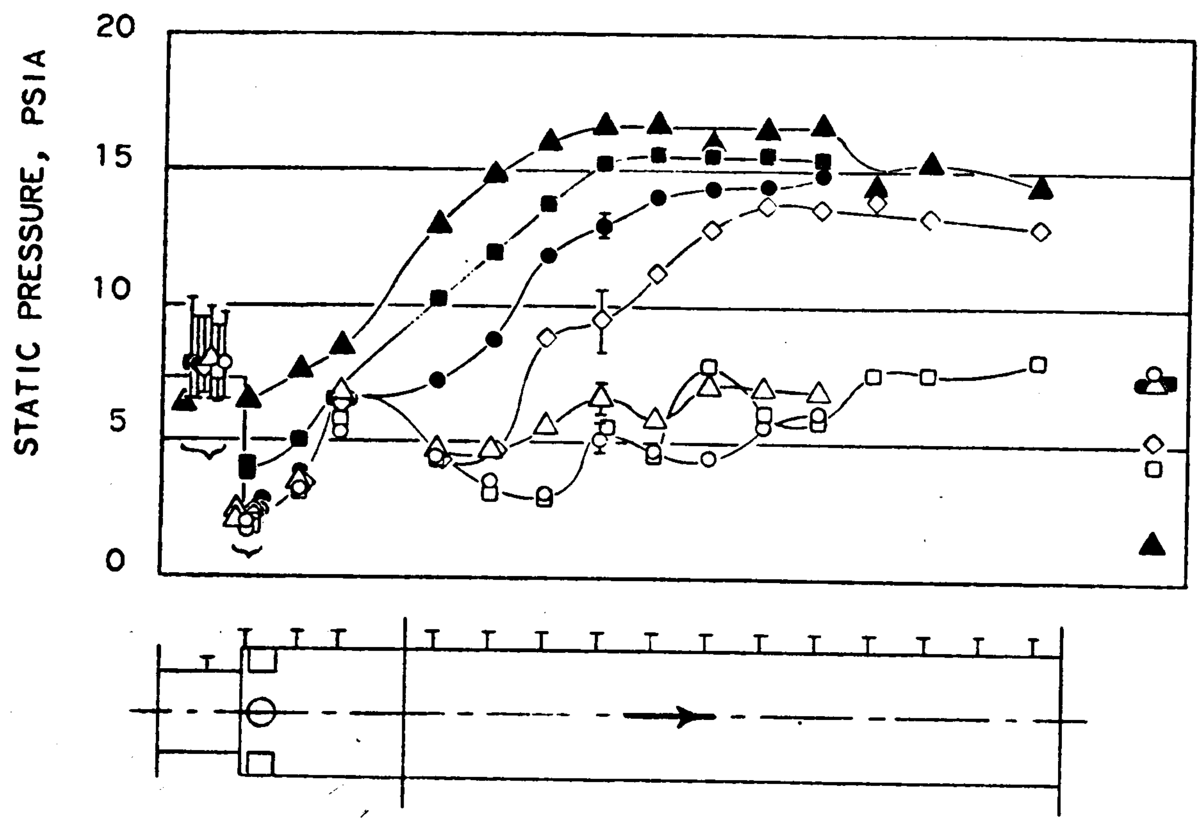


Figure 10 - Supersonic Combustor Models (Schematic).



SYMBOL	H_{T2}	ER
○	606	0
□	621	.20
△	606	.29
◇	645	.40
•	689	.63
■	770	.76
▲	766	.88
●	759	1.00

Figure 11 - Wall Static Pressure, Configuration AEN



SYMBOL	H_{T2}	ER
○	1233	.33
□	1229	.43
△	1296	.51
◇	1279	.59
●	1233	.71
■	1222	.88
▲	1291	1.26

Figure 12 - **Wall Static Pressure, Configuration AEN**

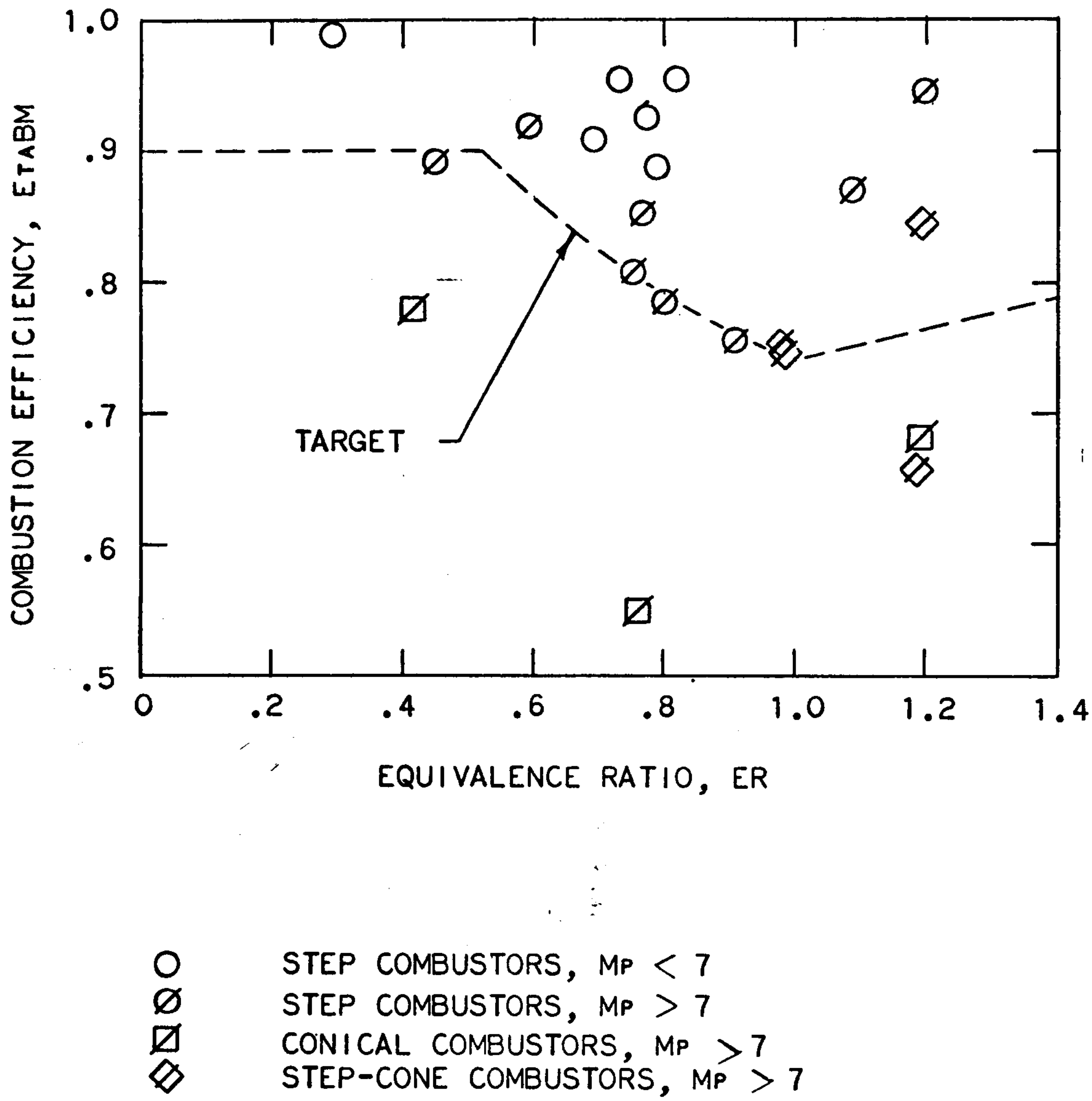


Figure 13 - **Combustion Efficiency VS Equivalence Ratio**

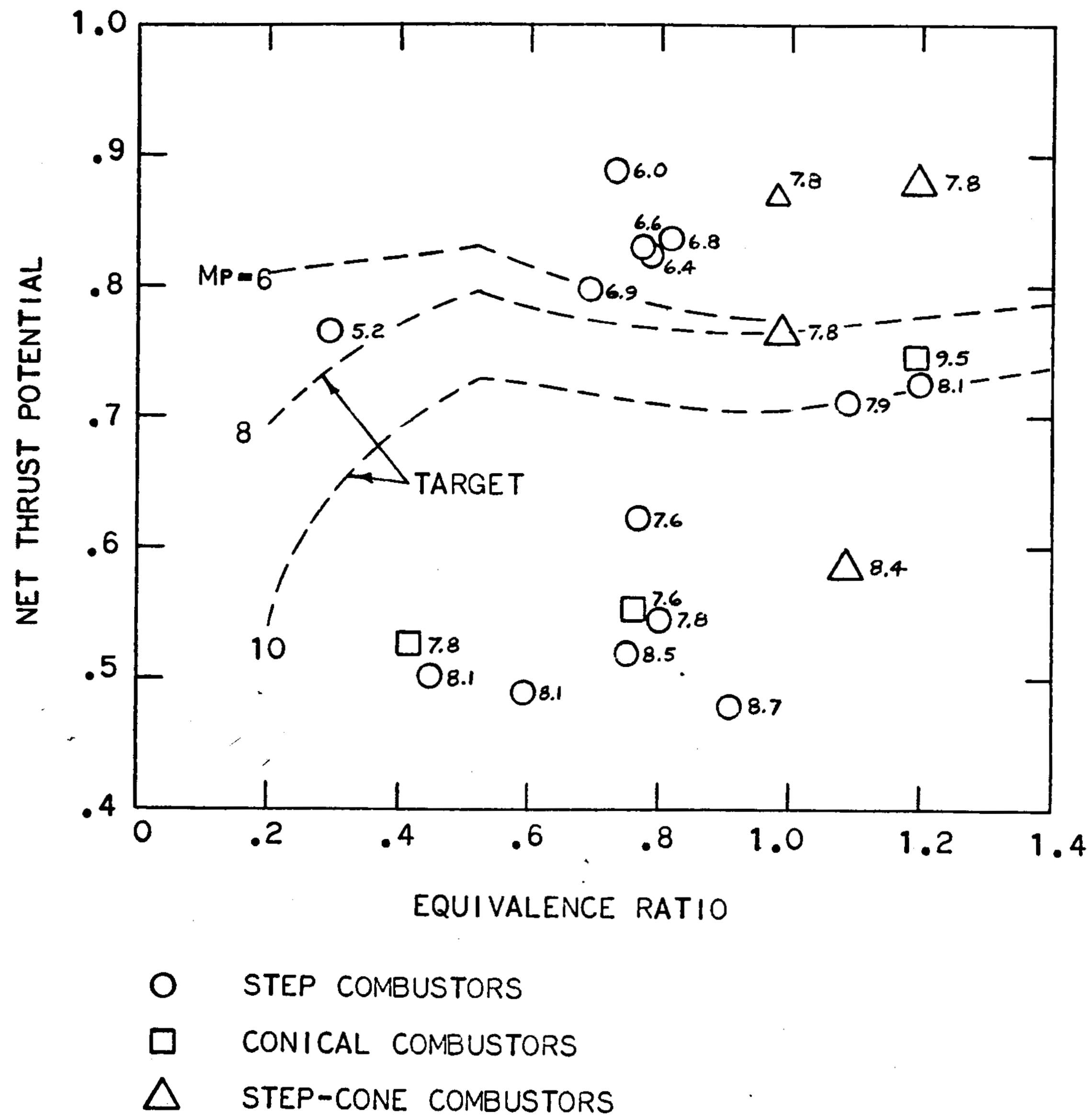
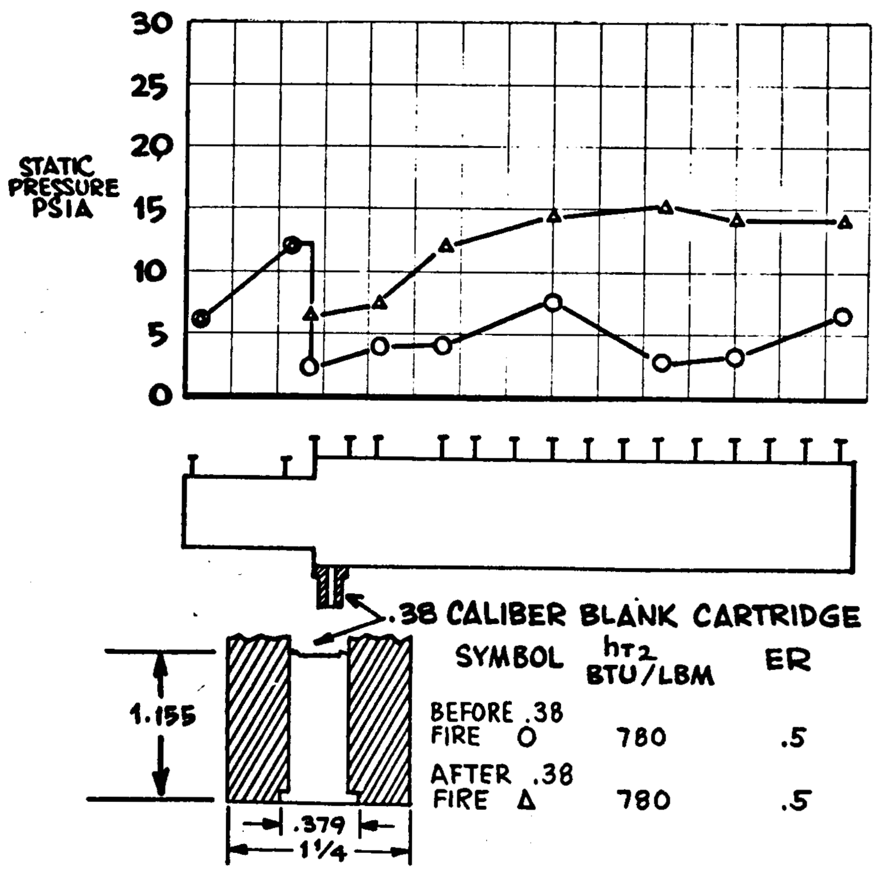


Figure 14 - **Thrust Potential VS Equivalence Ratio**



Step Combustor Ignition Test, Configuration A(G E38)O

Figure 15:

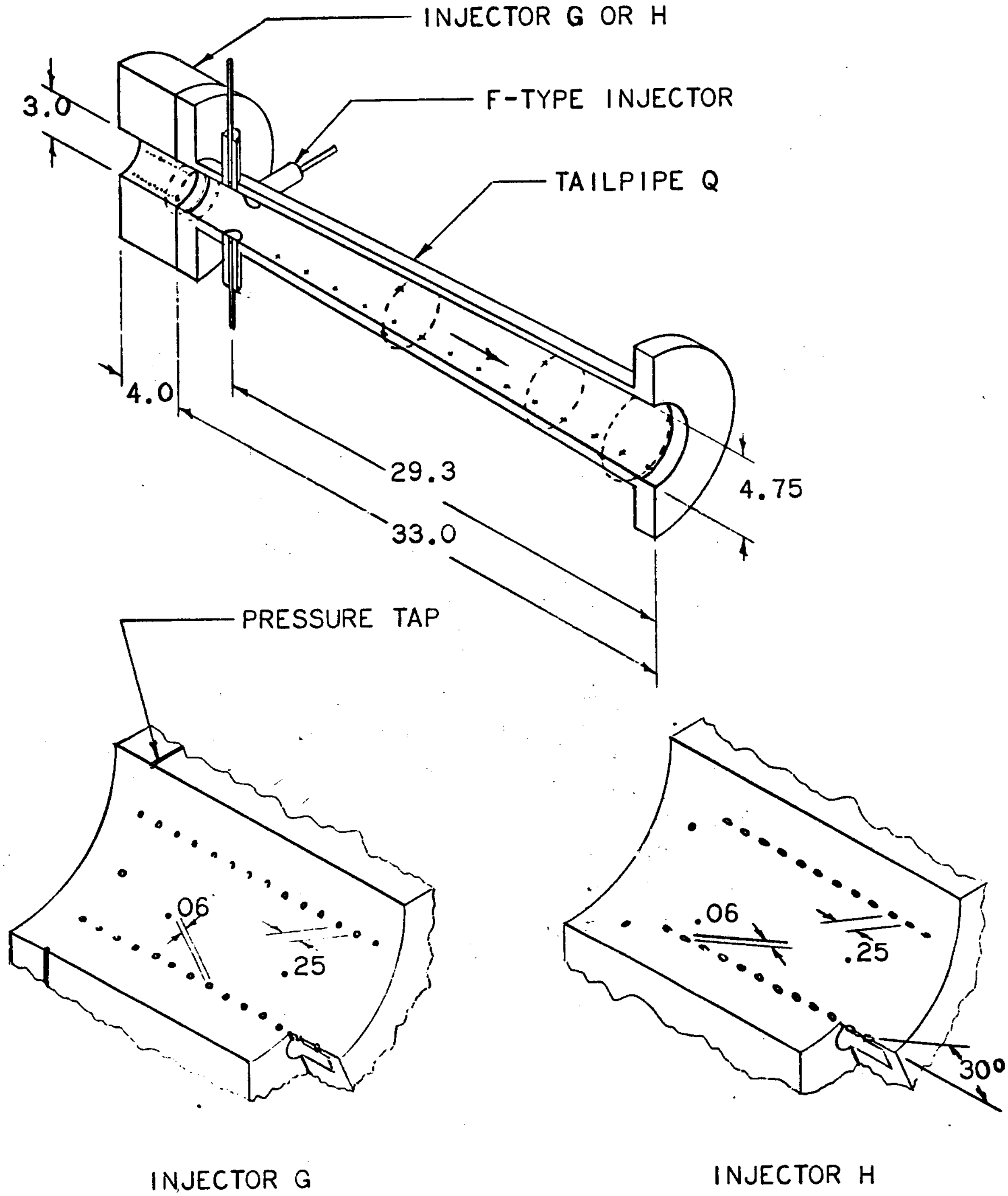


Figure 16 - Supersonic Combustor Models (Schematic).

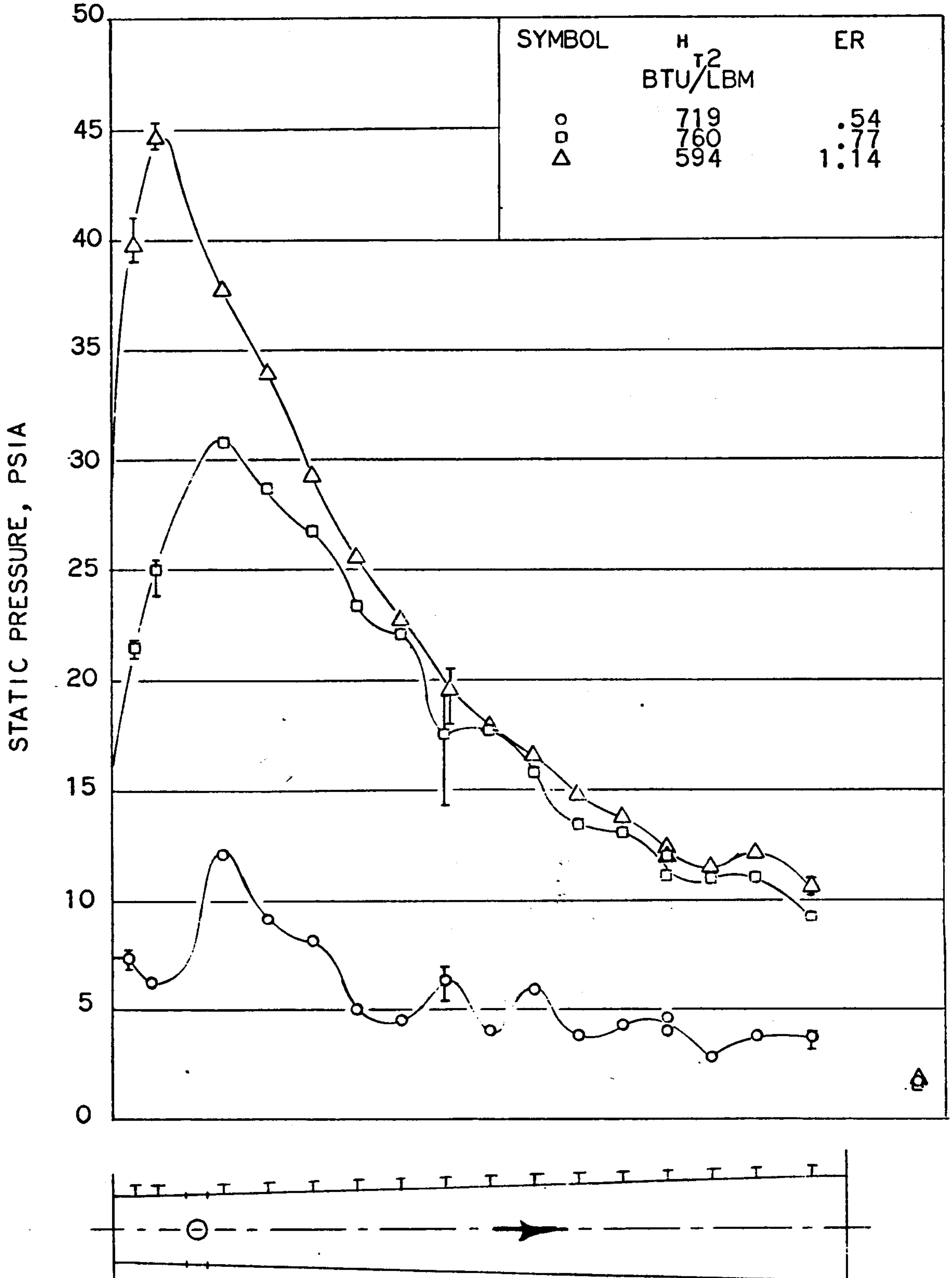
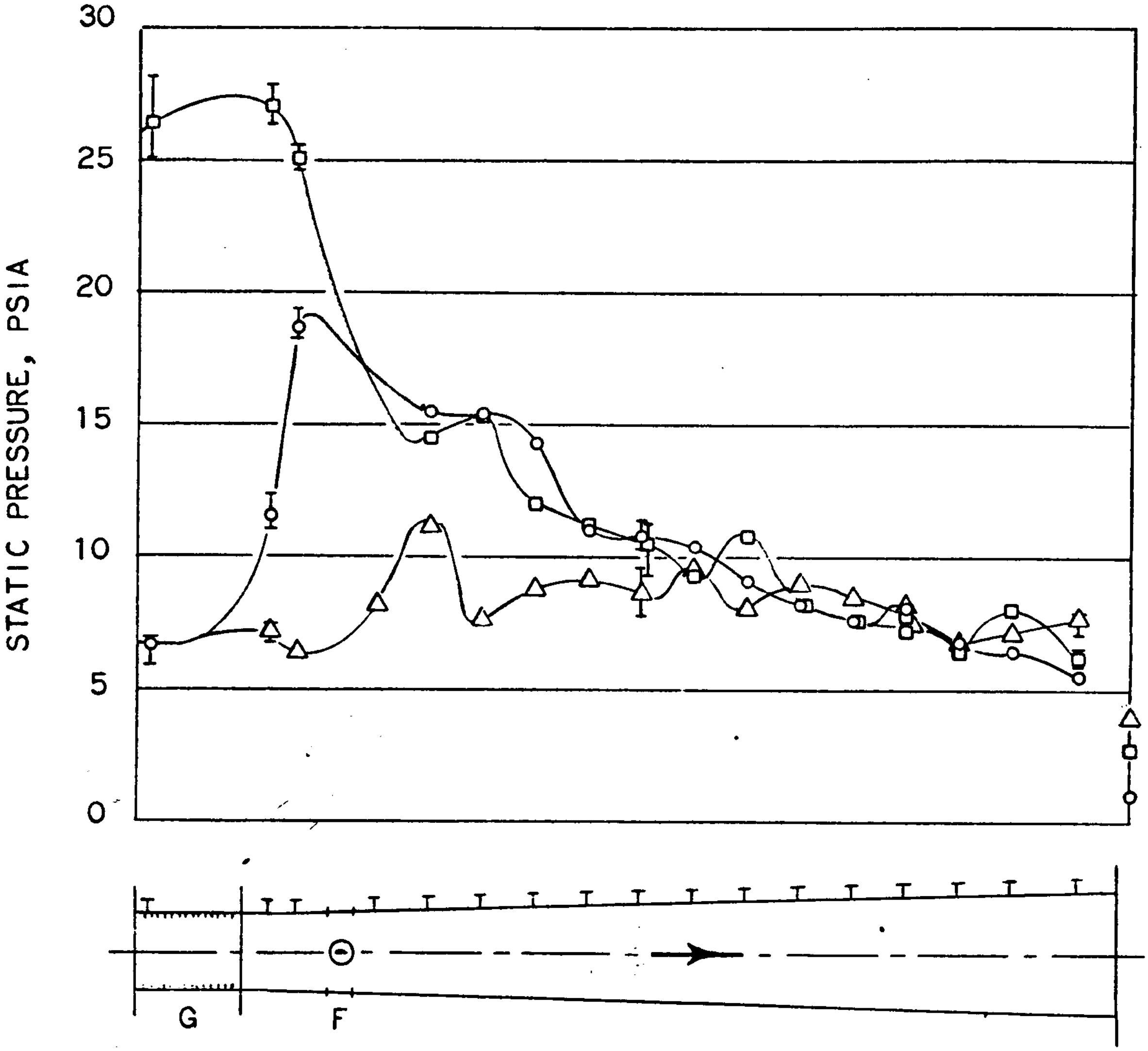


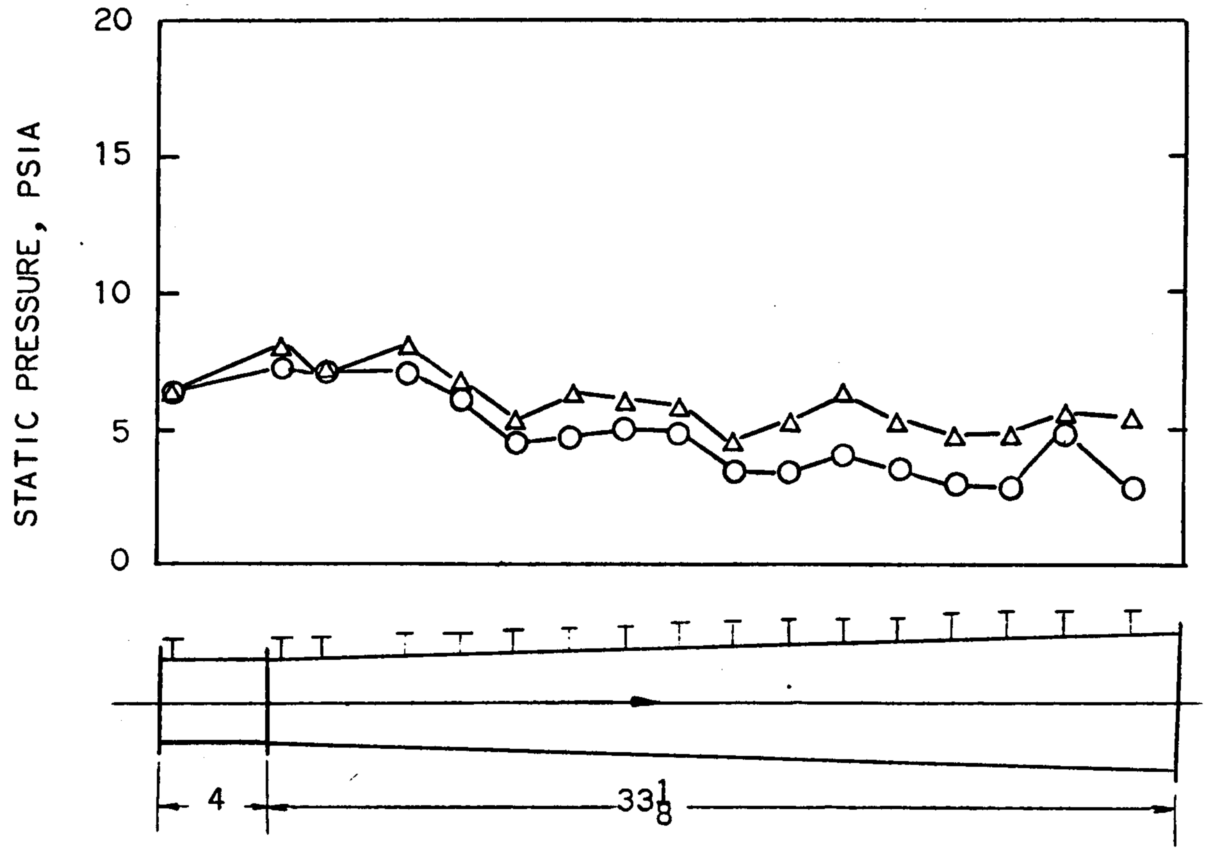
Figure 17 - Wall Static Pressures, Configuration AFQ



SYMBOL	H_{T2} BTU/LBM	ER G	ER F	ER TOTAL
○	1255	.42	.56	.98
□	1262	.83	0	.83
△	1268	0	.96	.96

Figure 18 - Wall Static Pressures, Configuration AGFQ

[redacted]



SYMBOL	H_{T2}	ER	RDG
○	1433	.170	4
△	1471	.712	6C

Figure 19 - Wall Static Pressure, Configuration BHQ

[redacted]

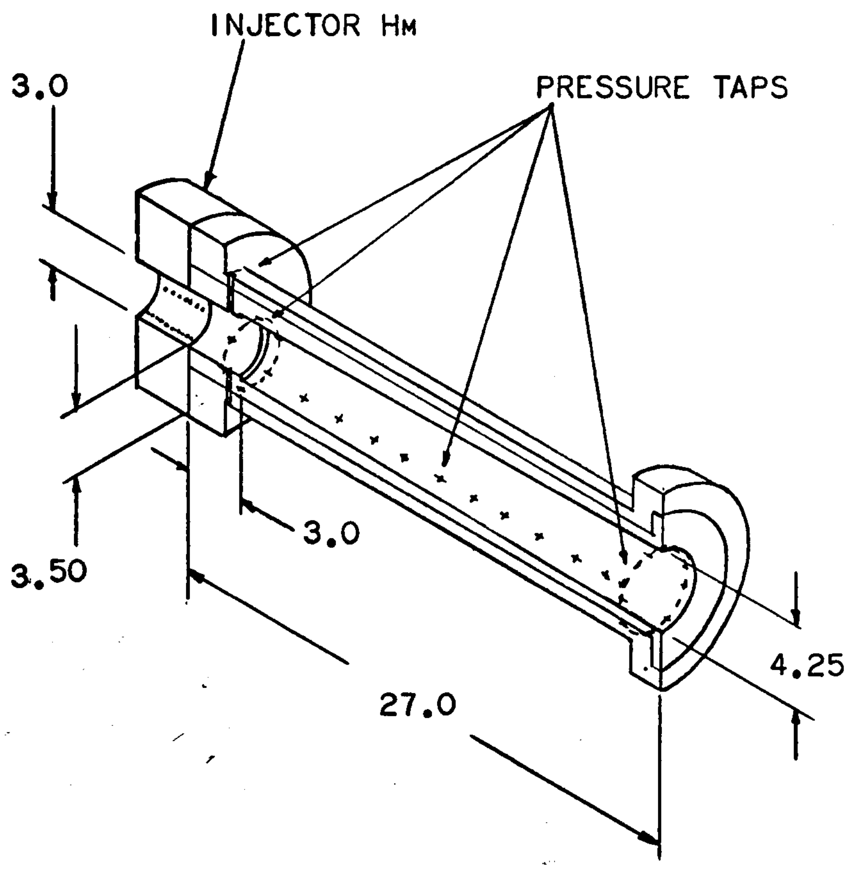


Figure 20 **- Tailpipe R (Schematic).**

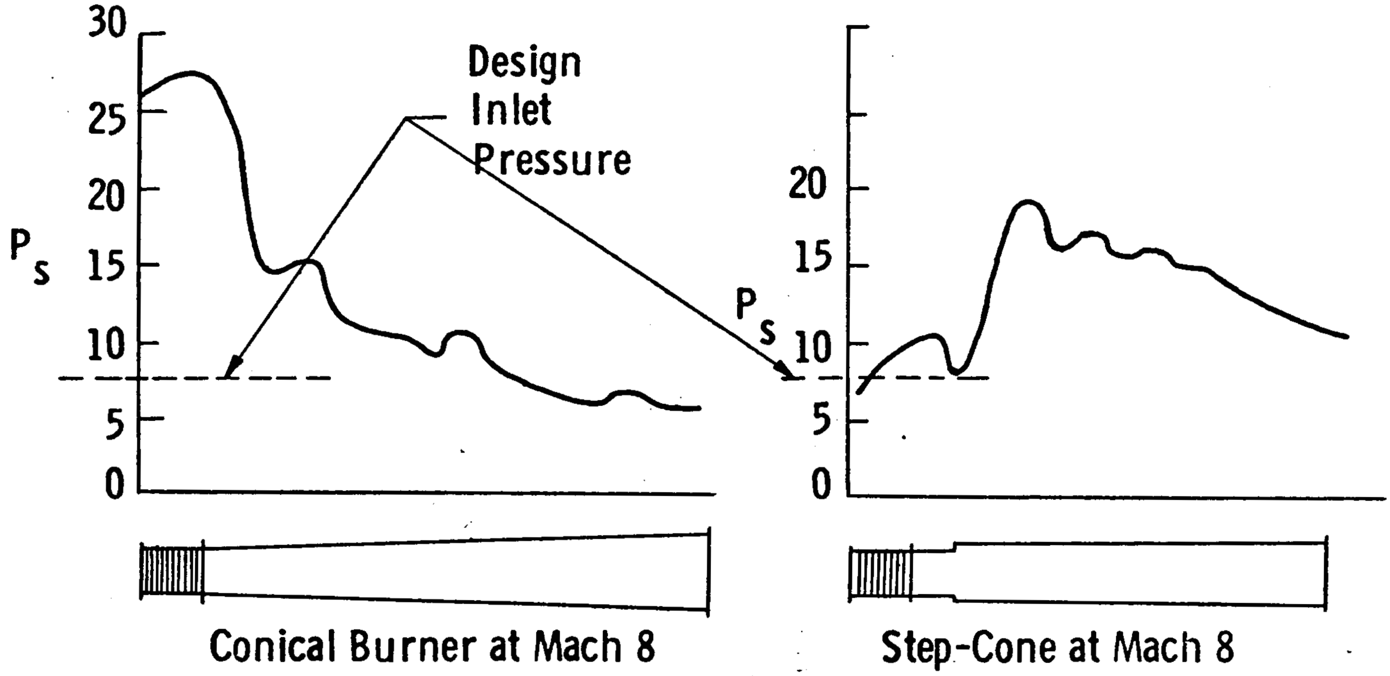
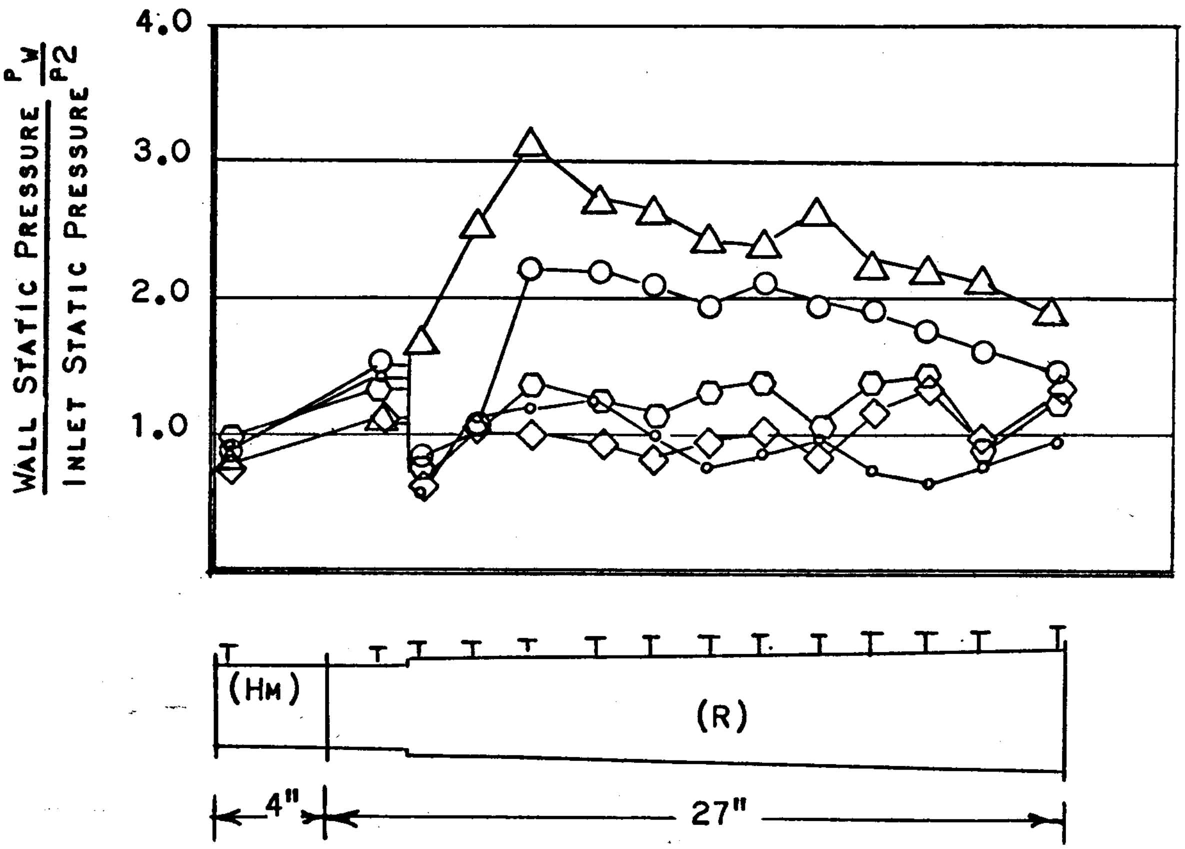
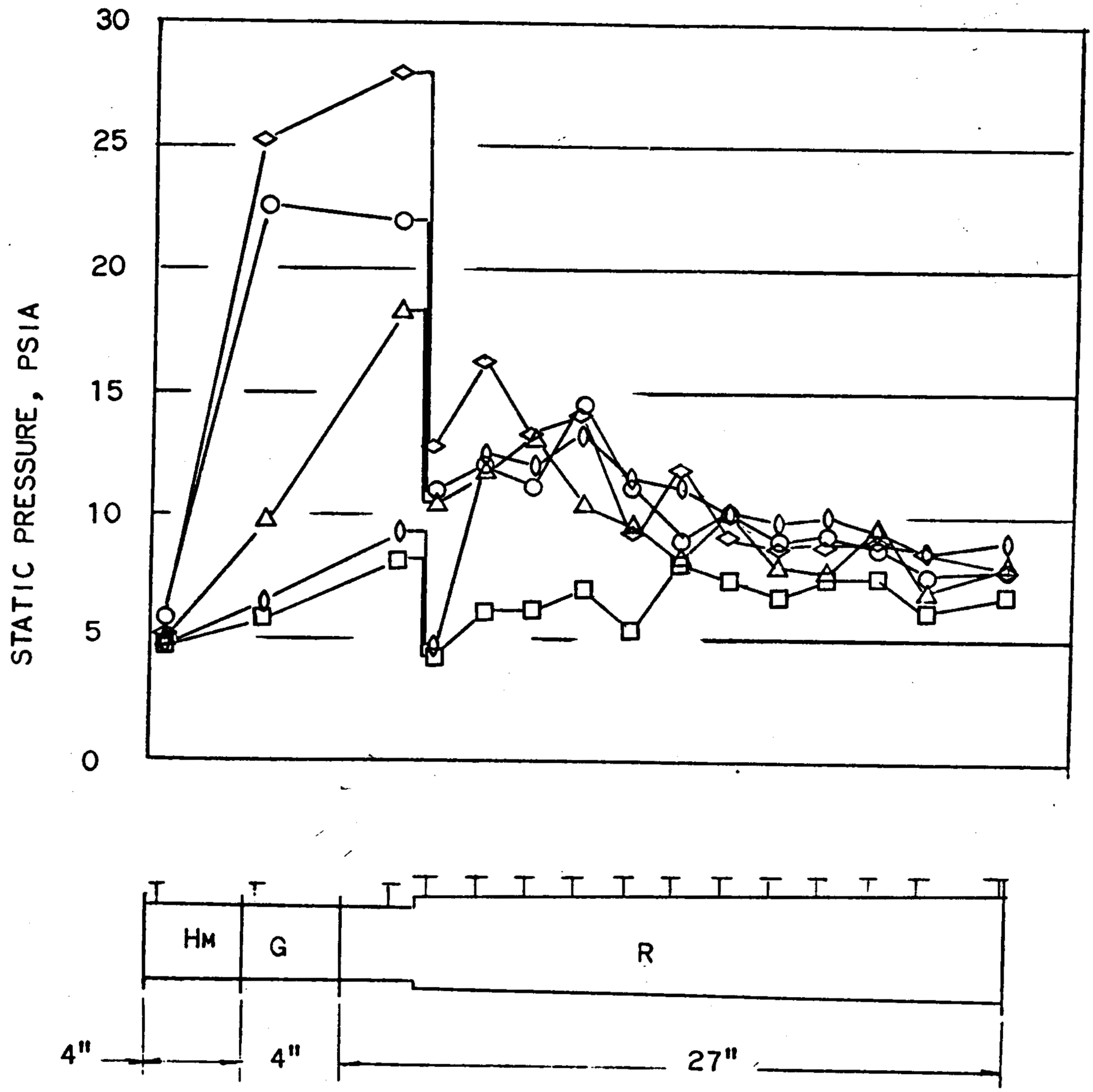


Figure 21: Effect of Step on Combustion-Induced Inlet Unstarts



SYMBOL	H_{T2}	M_P	ER	P_{S2}	RUN
○	1236	7.75	1.03	7.34	206-7A
△	1273	7.94	.98	5.88	212-3A
◇	1427	8.44	1.09	5.45	210-1D
⬡	1512	8.71	1.07	3.98	210-5C
°	1649	9.12	1.17	3.37	210-7D

Figure 22 - Normalized Static Pressure, Configuration BHmR



SYMBOL	H_{T2}	MP	ER (HM)	ER (G)
○	1575	8.89	.199	.925
△	1575	8.89	.204	.775
◇	1580	8.92	.200	.375
0	1625	9.04	.751	.187
□	1575	8.89	.973	.187

Figure 23 - Wall Static Pressures, Configuration BHmGR

TWO-DIMENSIONAL TEST COMBUSTOR

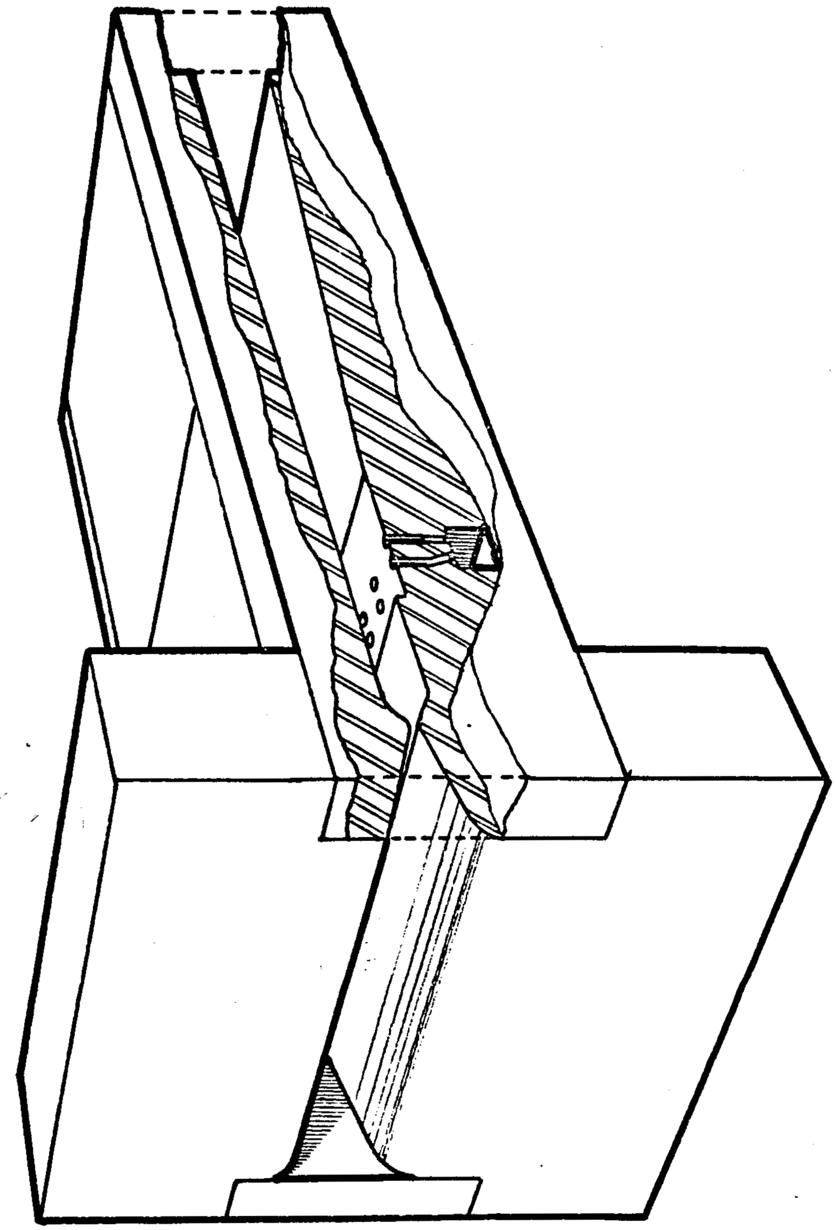


Figure 24:

COMPARISON OF STATIC PRESSURE TRACES, 2-D VS. CONICAL BURNER

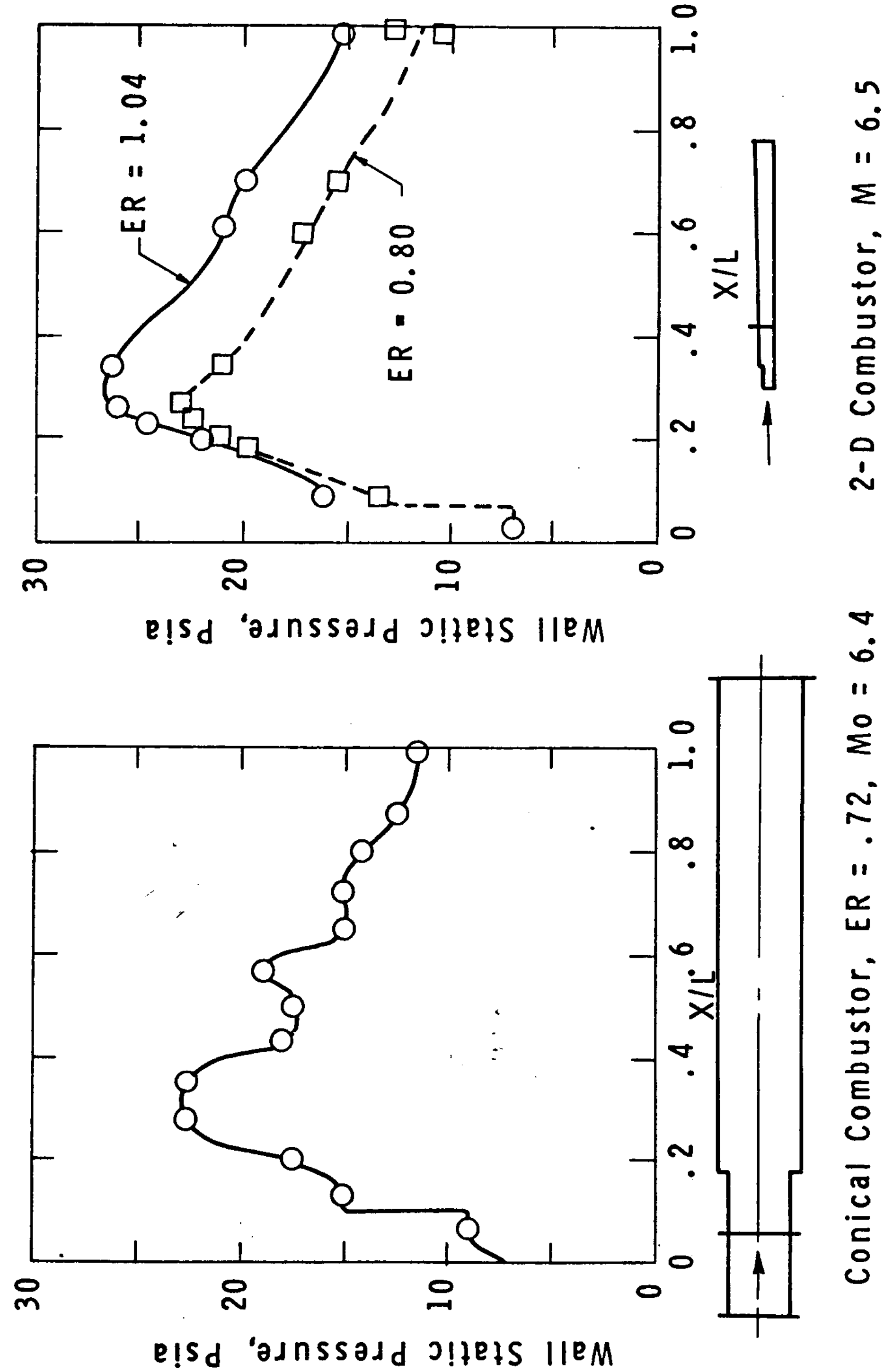


Figure 25:

AREA RATIO COMPARISON OF 2-D BURNERS WITH

M = 8 BASELINE ENGINE

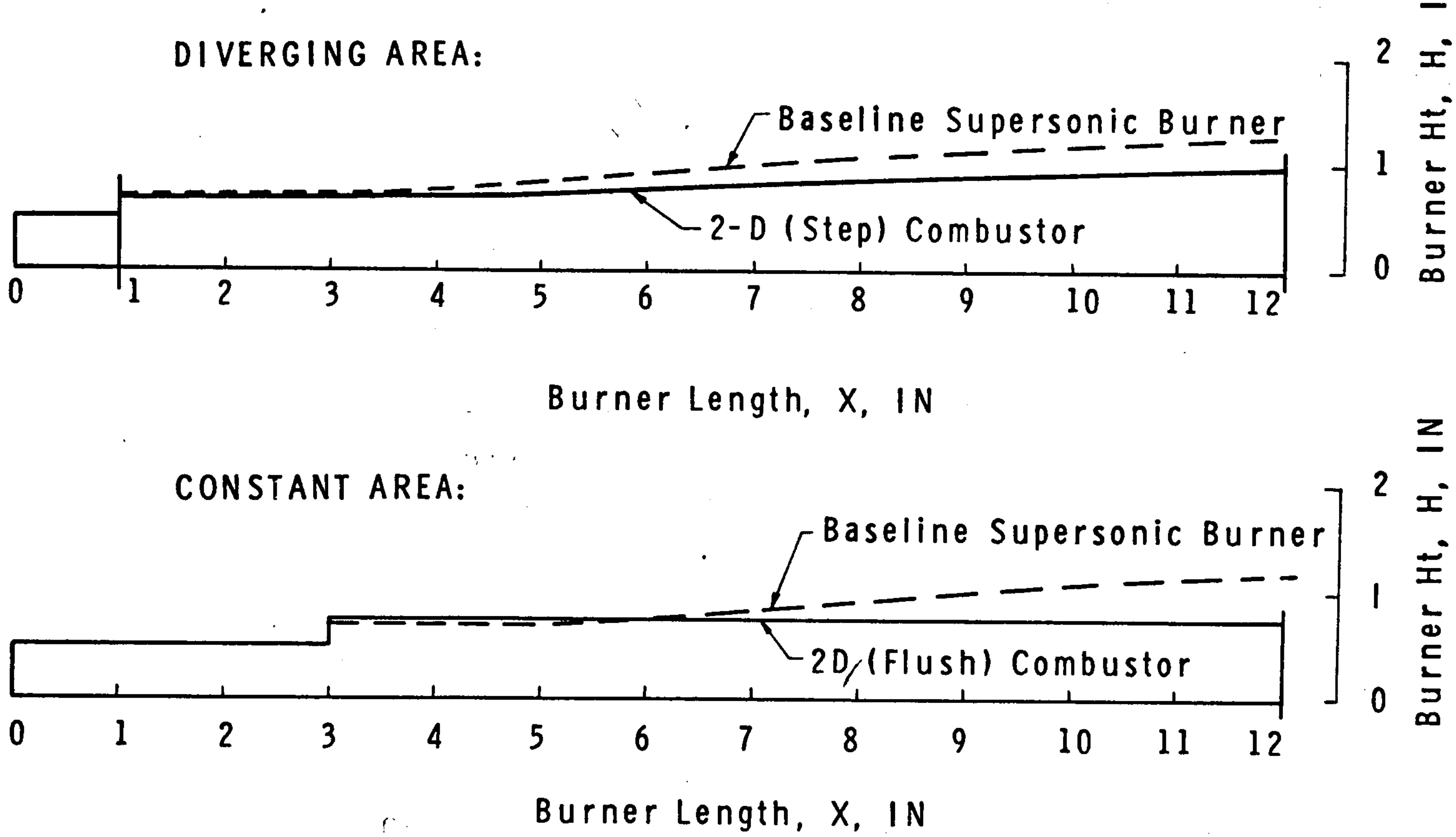


Figure 26:

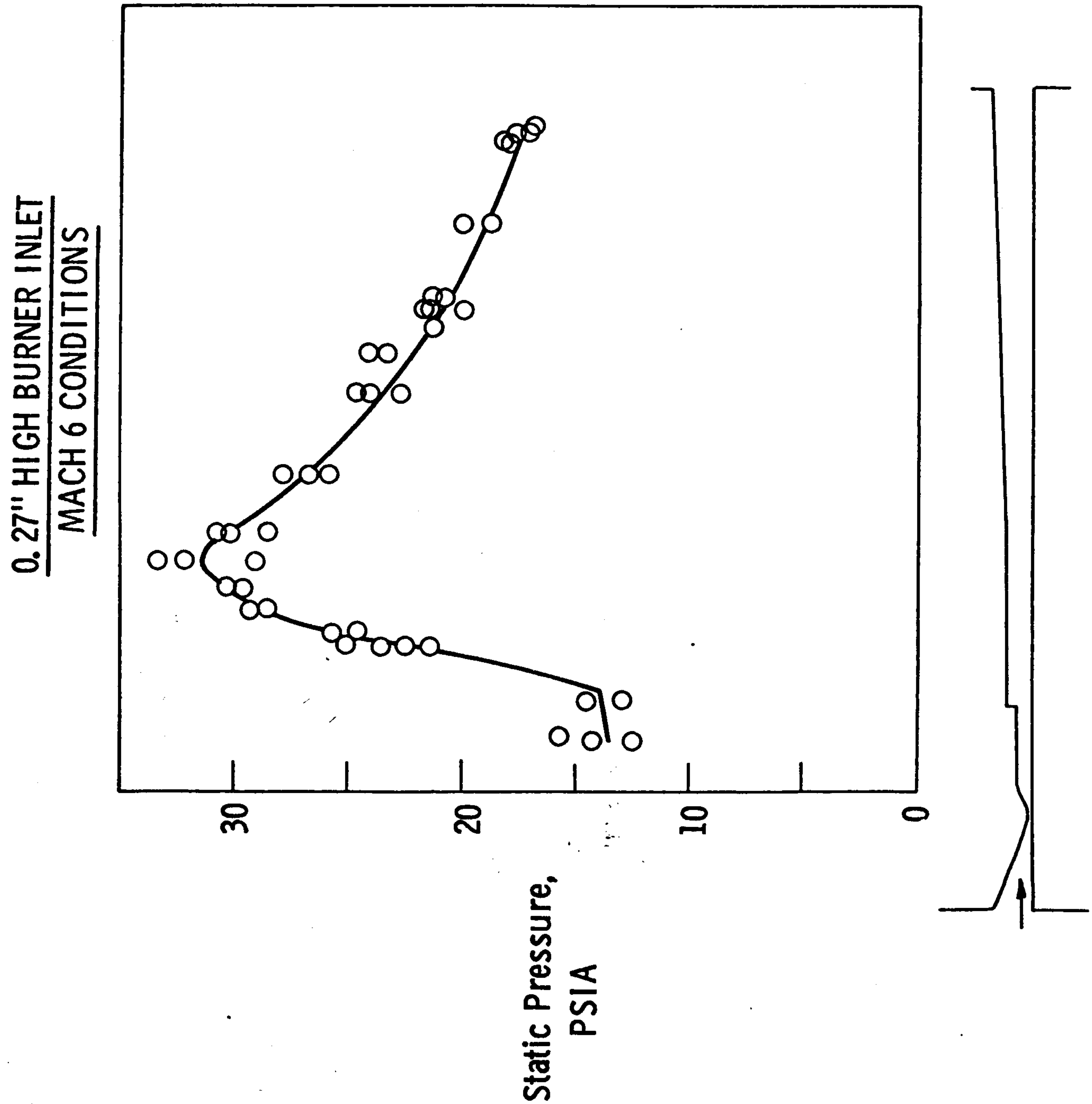

0.27" HIGH BURNER INLET
MACH 6 CONDITIONS
Static Pressure,
PSIA
0
10
20
30

Figure 27:

MEASURED STABLE BURNING LIMITS

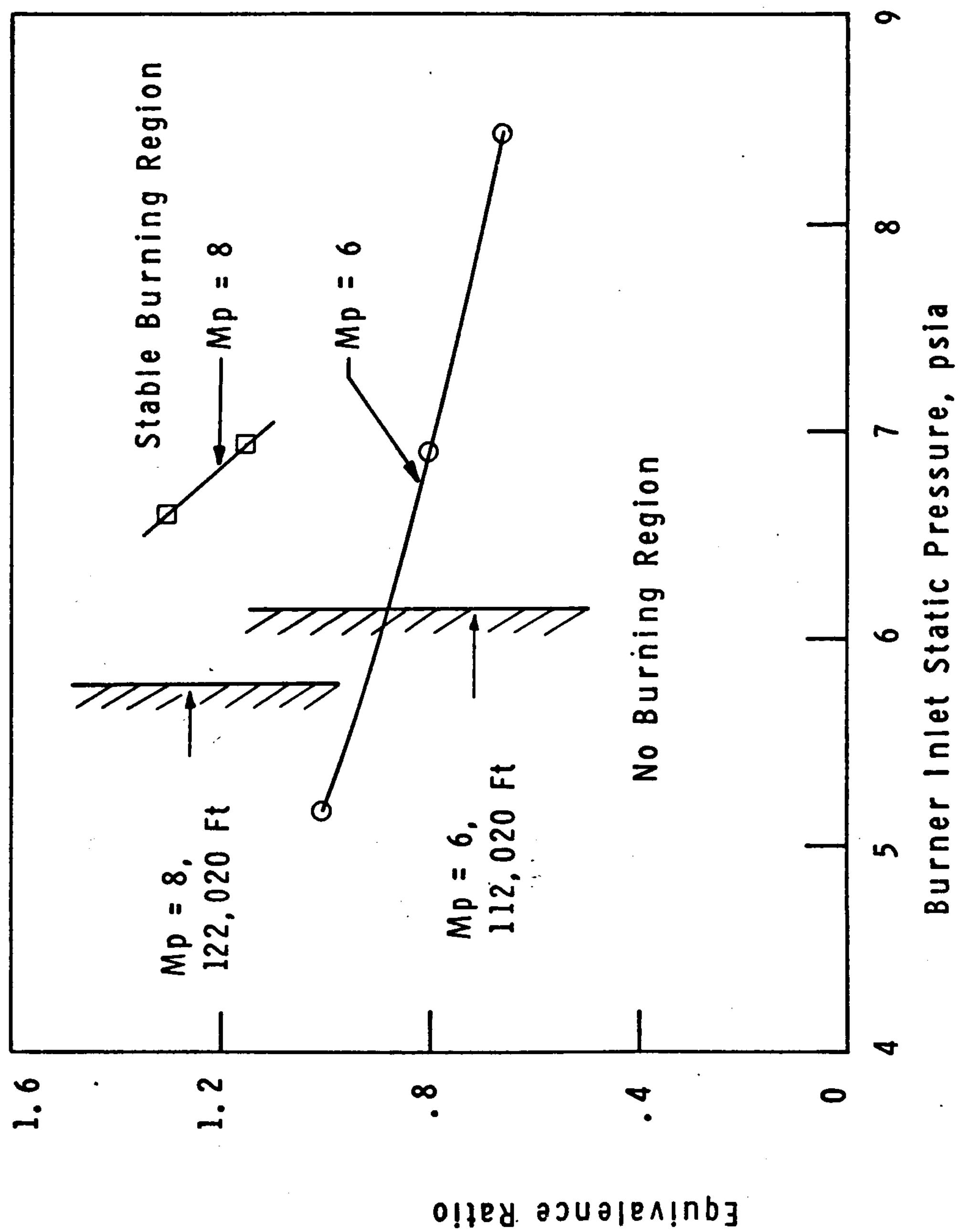
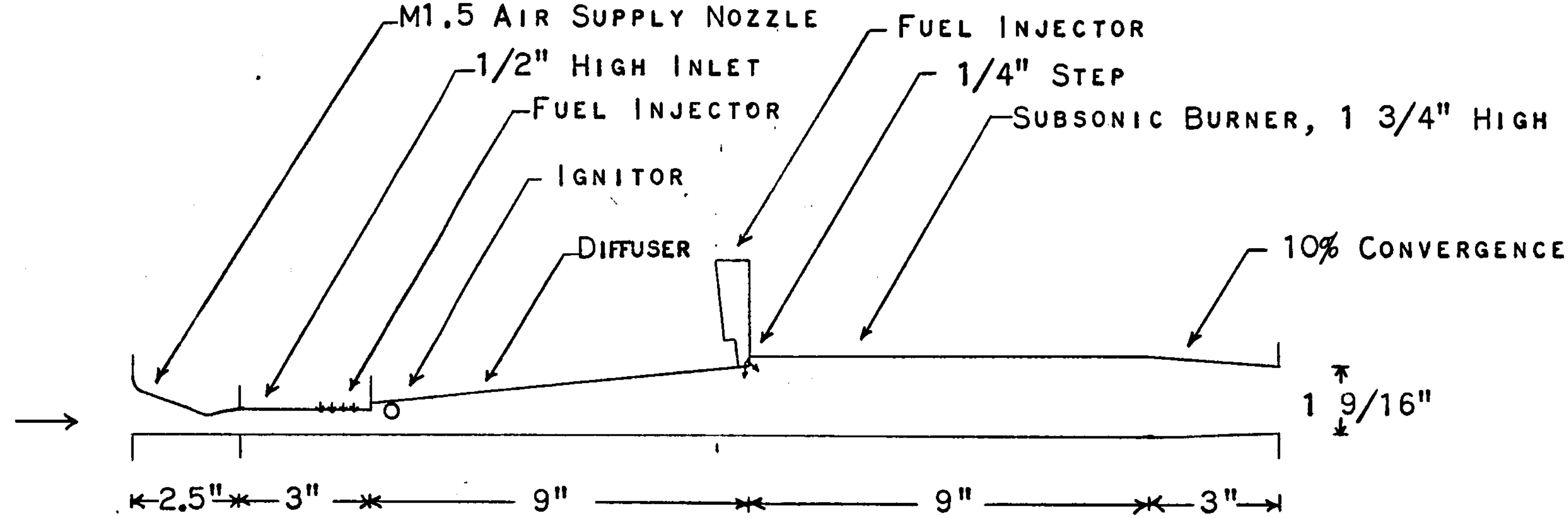


Figure 28



FLOW PATH FOR MACH 3 IGNITION TESTS

(Subsonic Burner Installed Downstream of Two-Dimensional Supersonic Burner Hardware)

FIGURE 29

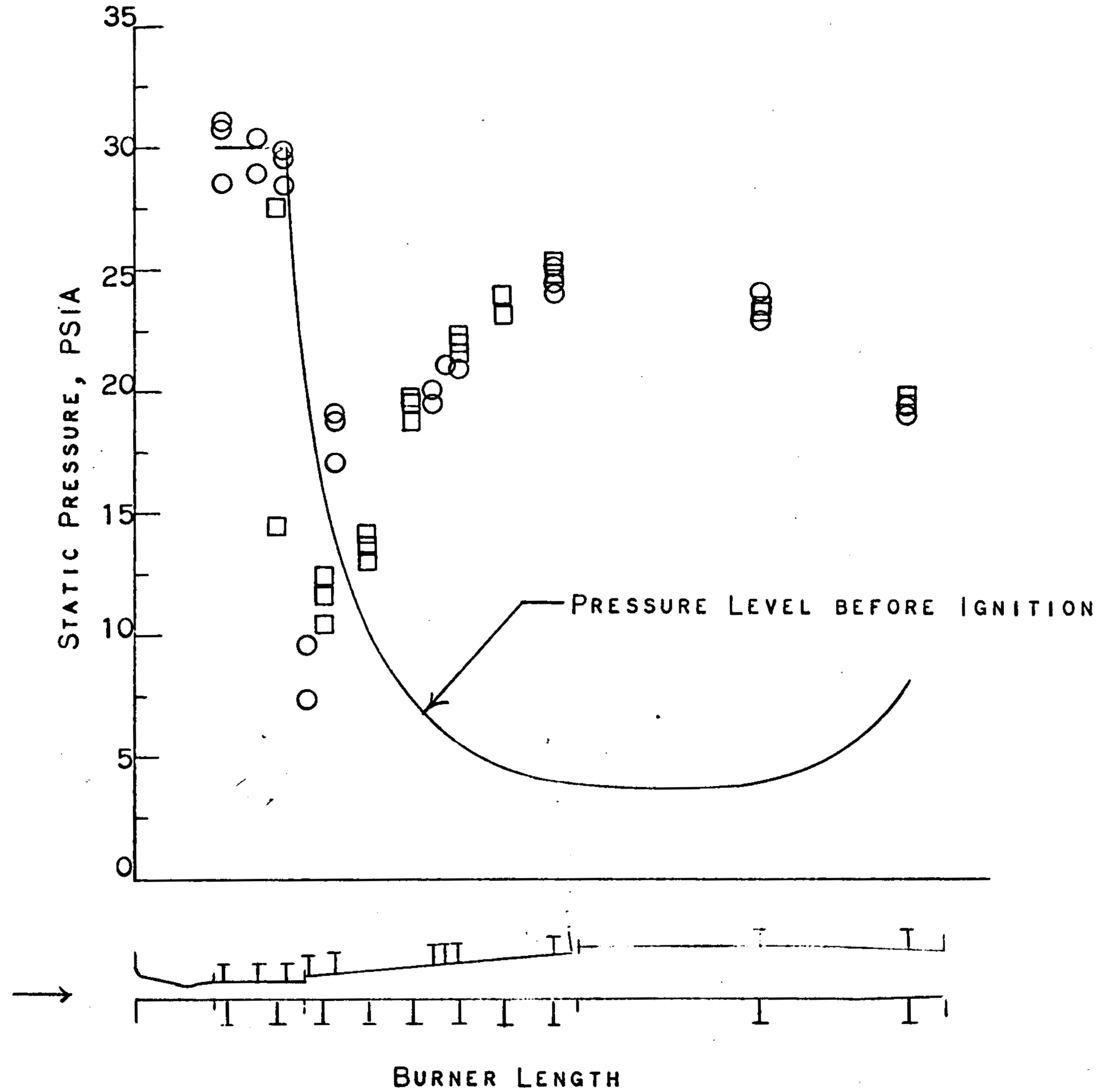
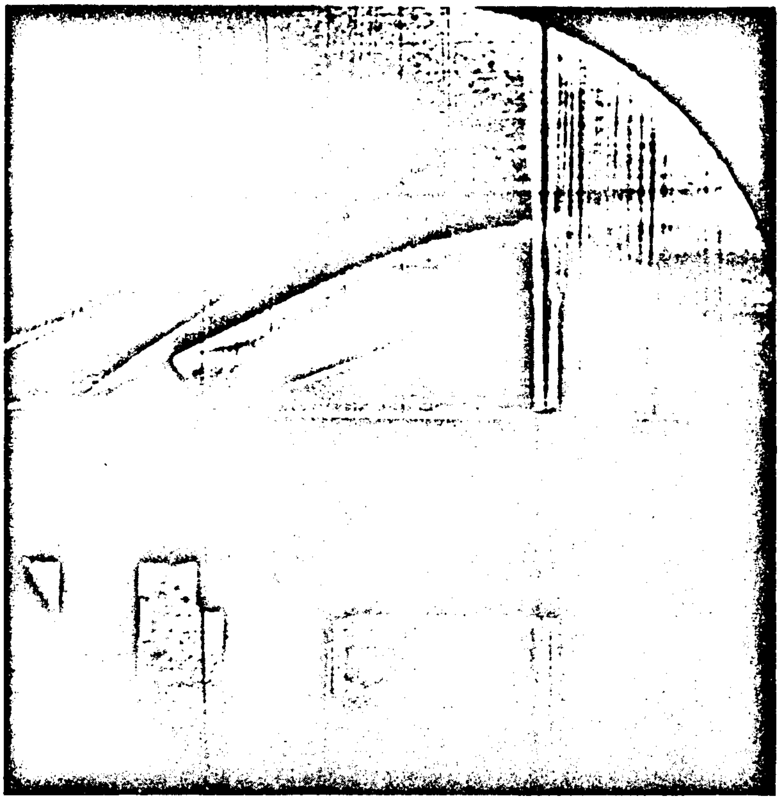
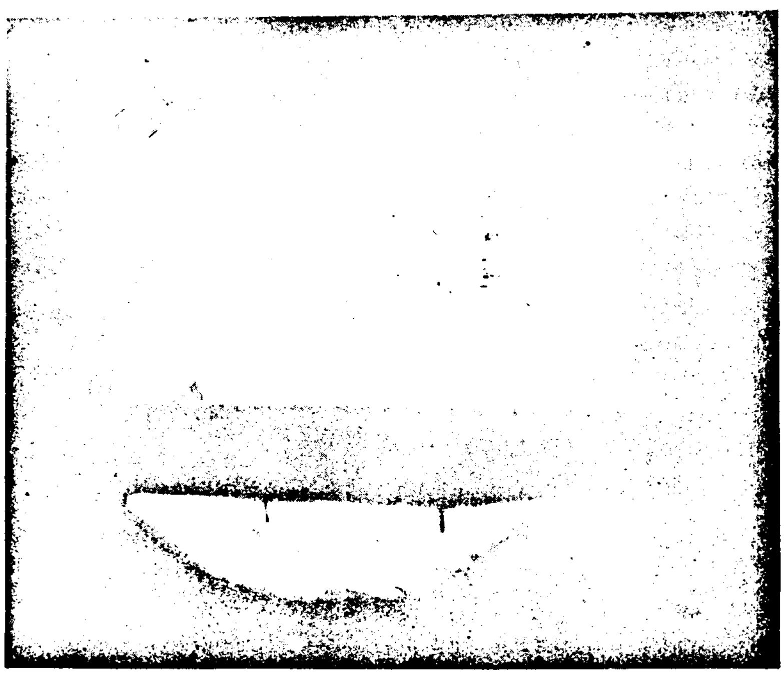


Figure 30: Pressure Level in Diffuser and Subsonic Burner After Ignition. (Equivalence Ratio = 0.5)



ARC TUNNEL TESTS



COLD FLOW TUNNEL TESTS

FIGURE 31: SCHLIEREN, CONFIGURATION VI, ARC TUNNEL TEST RUN NO. 70

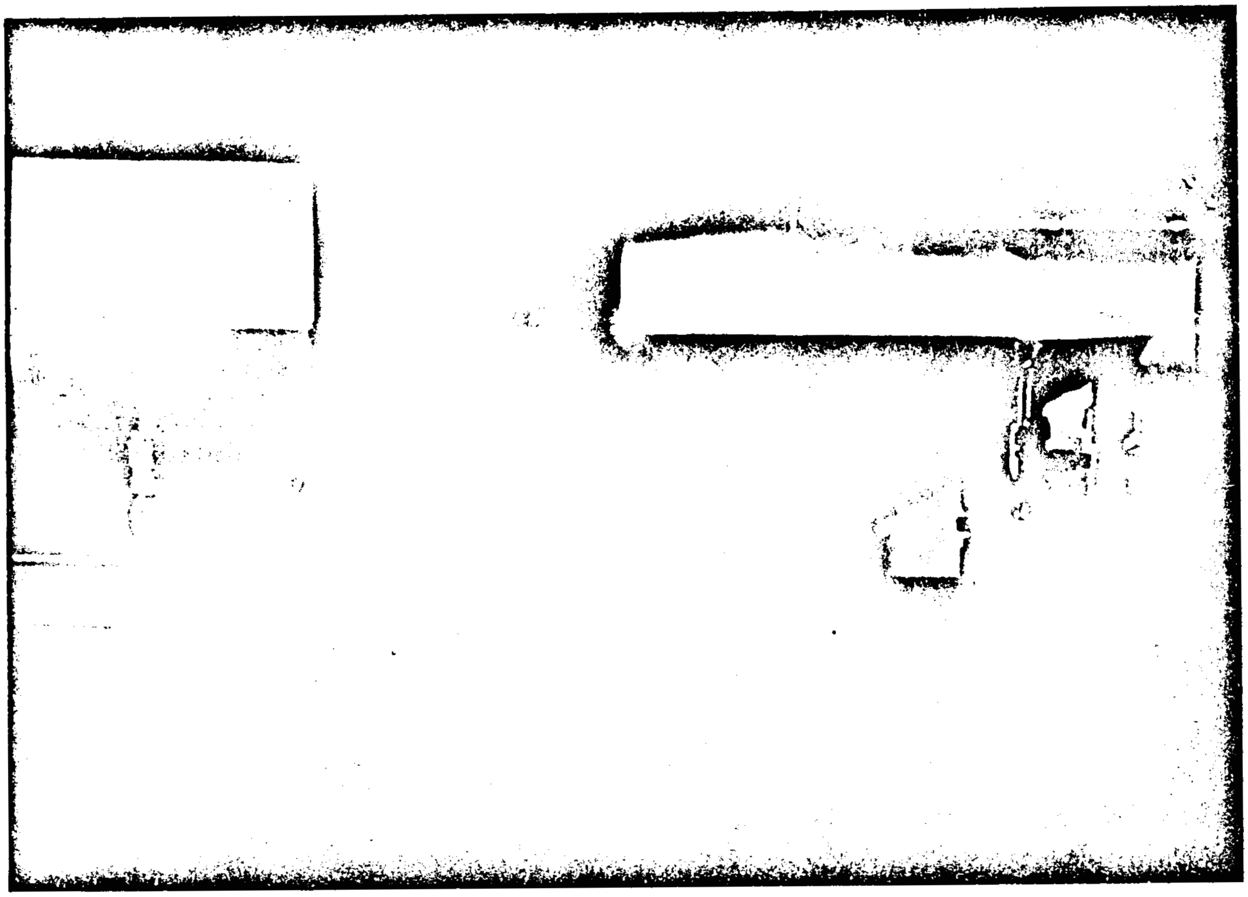


Figure 32. Photograph - Arc Tunnel Test Run No. 75, Hydrogen Combustion

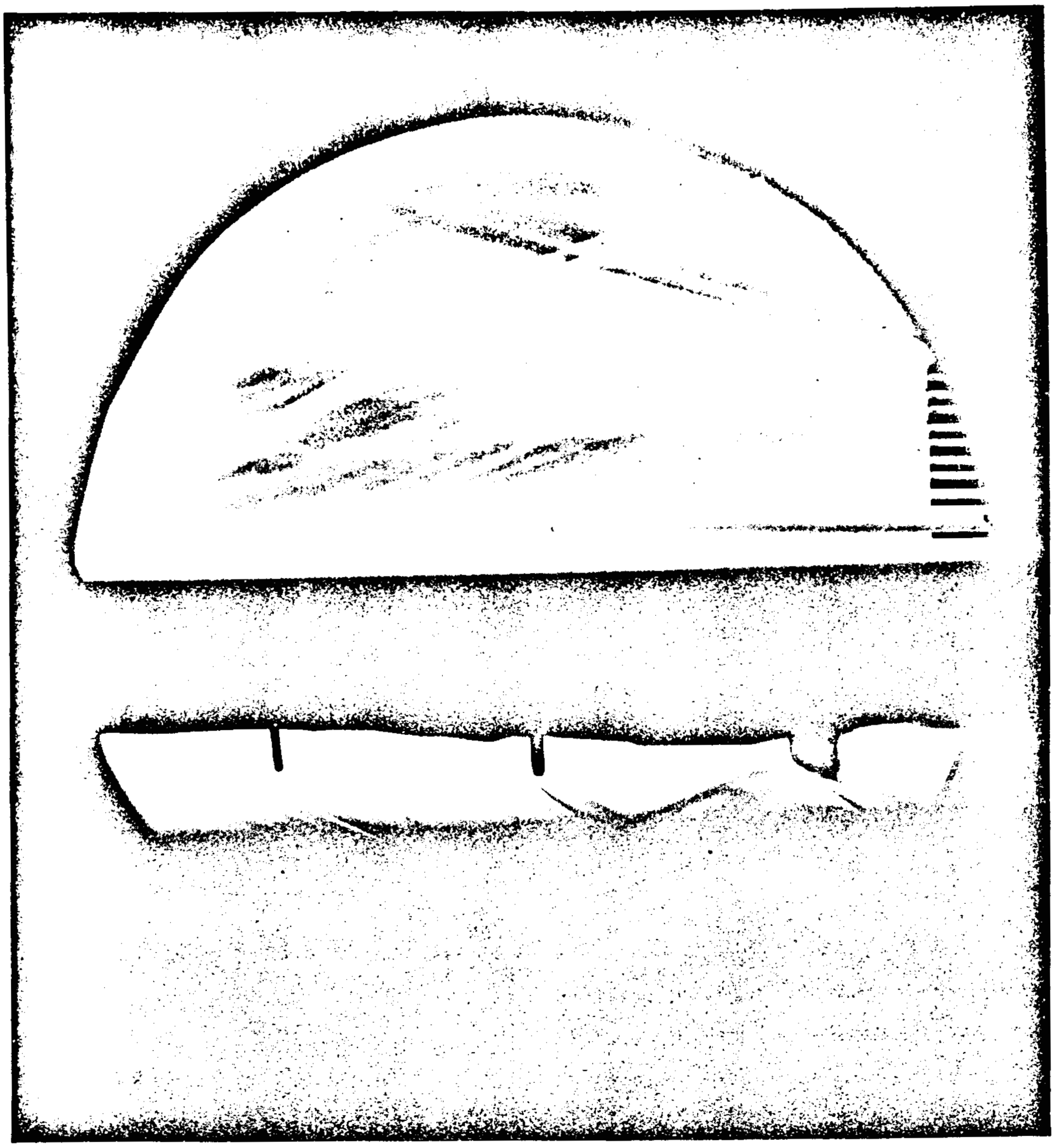
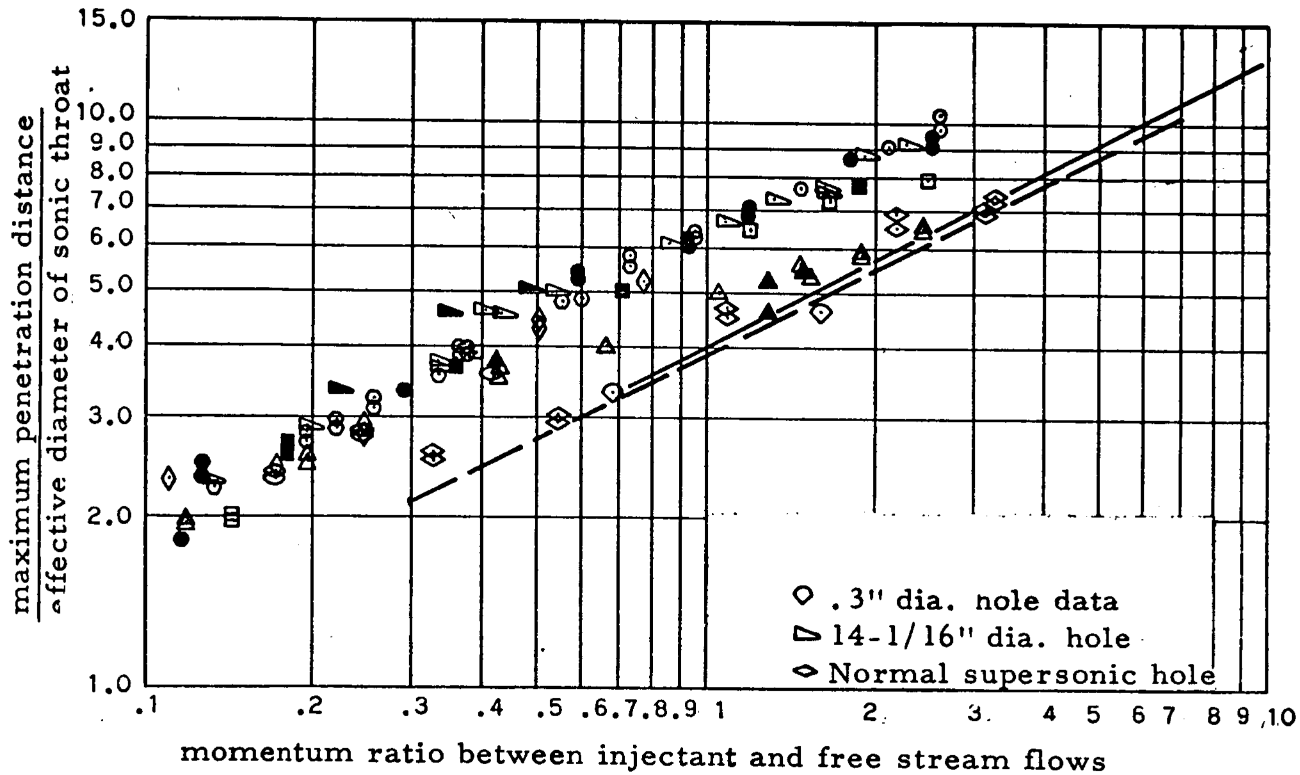


Figure 33 - Helium Penetration Using Distributed Continuous Light Source.



Correlation of Penetration Data

Figure 34:

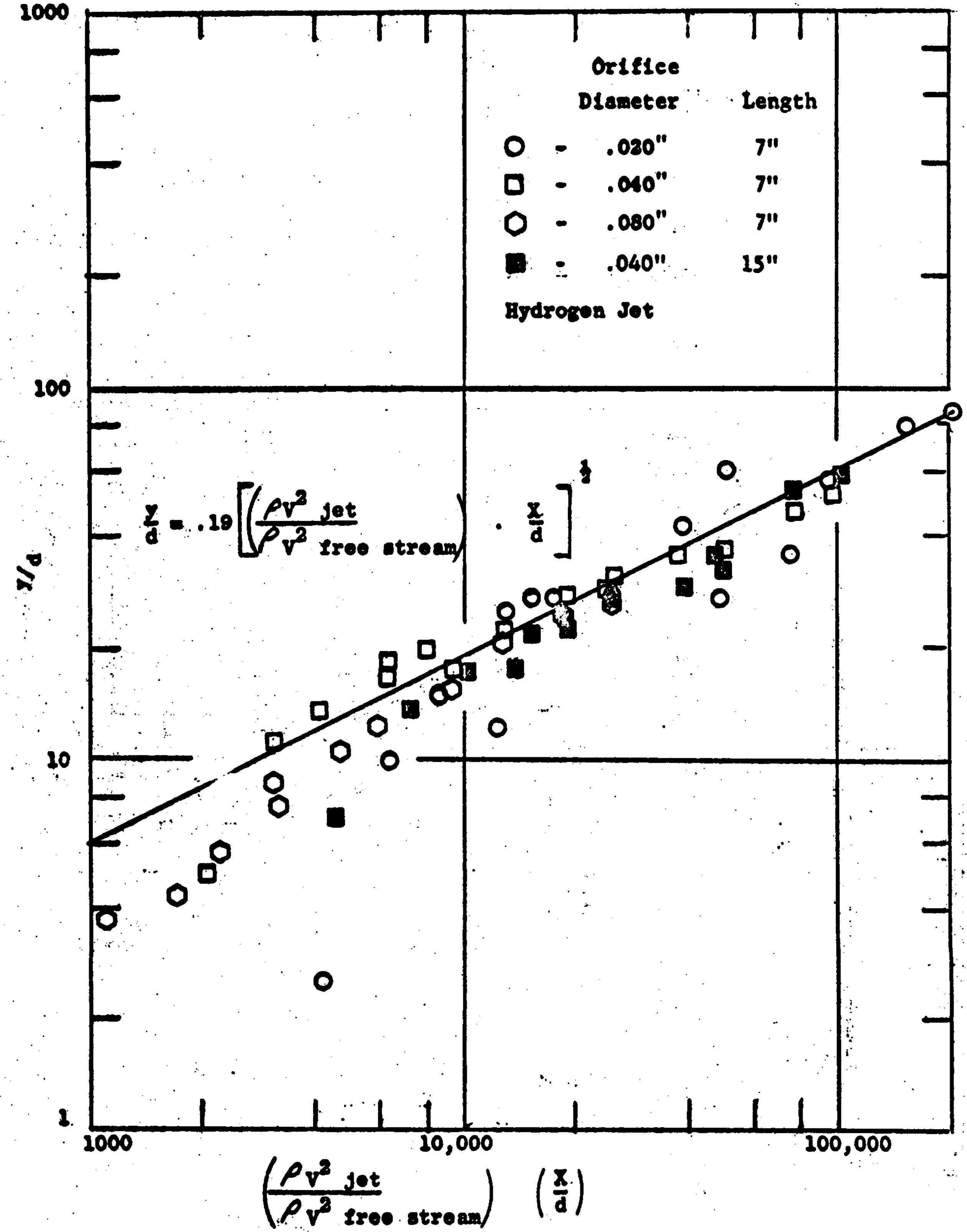


Figure 35- Jet Penetration Correlation for "Clean" Penetration Tests

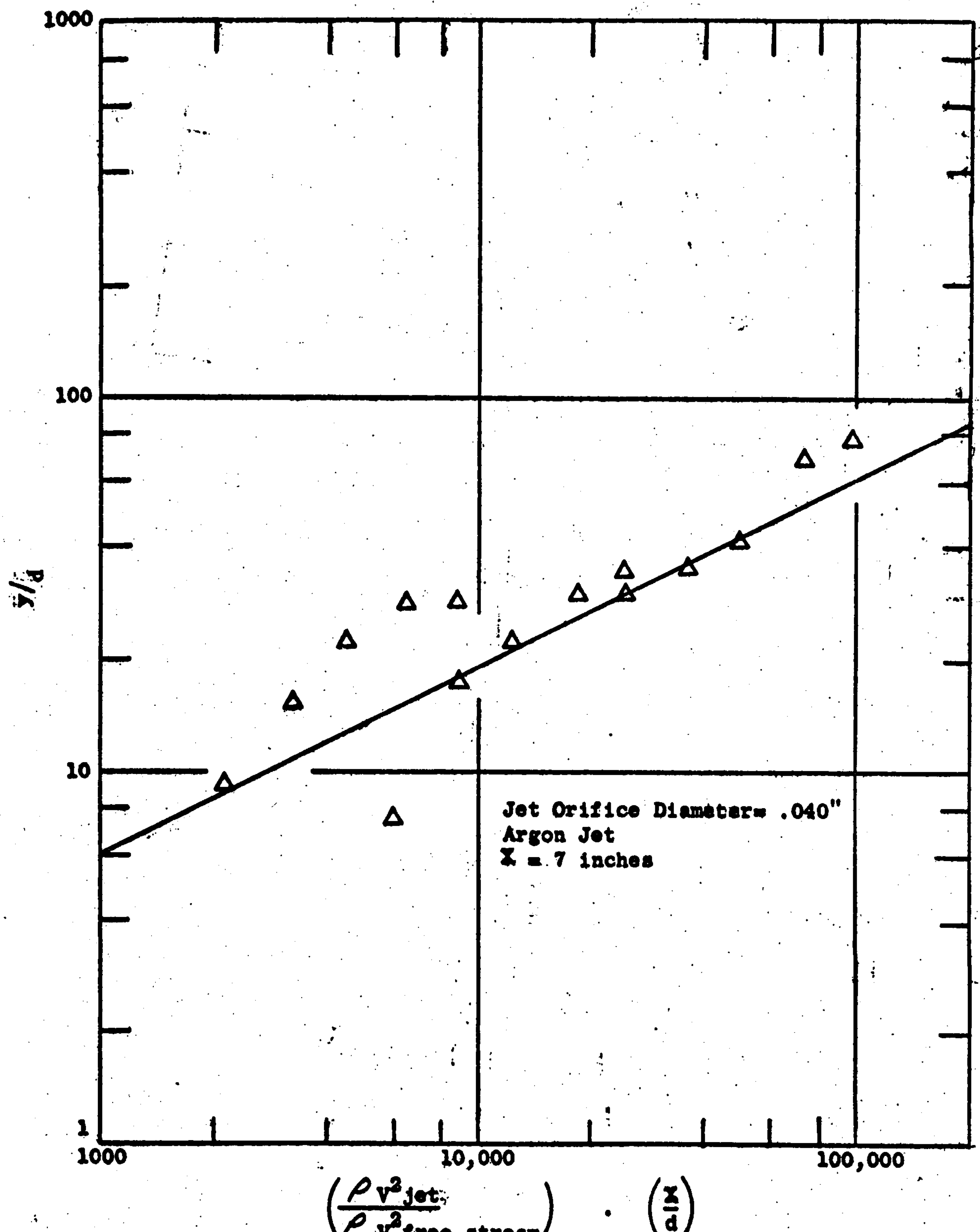


Figure 36 - Comparison of Argon Penetration Data to Hydrogen Jet Correlation

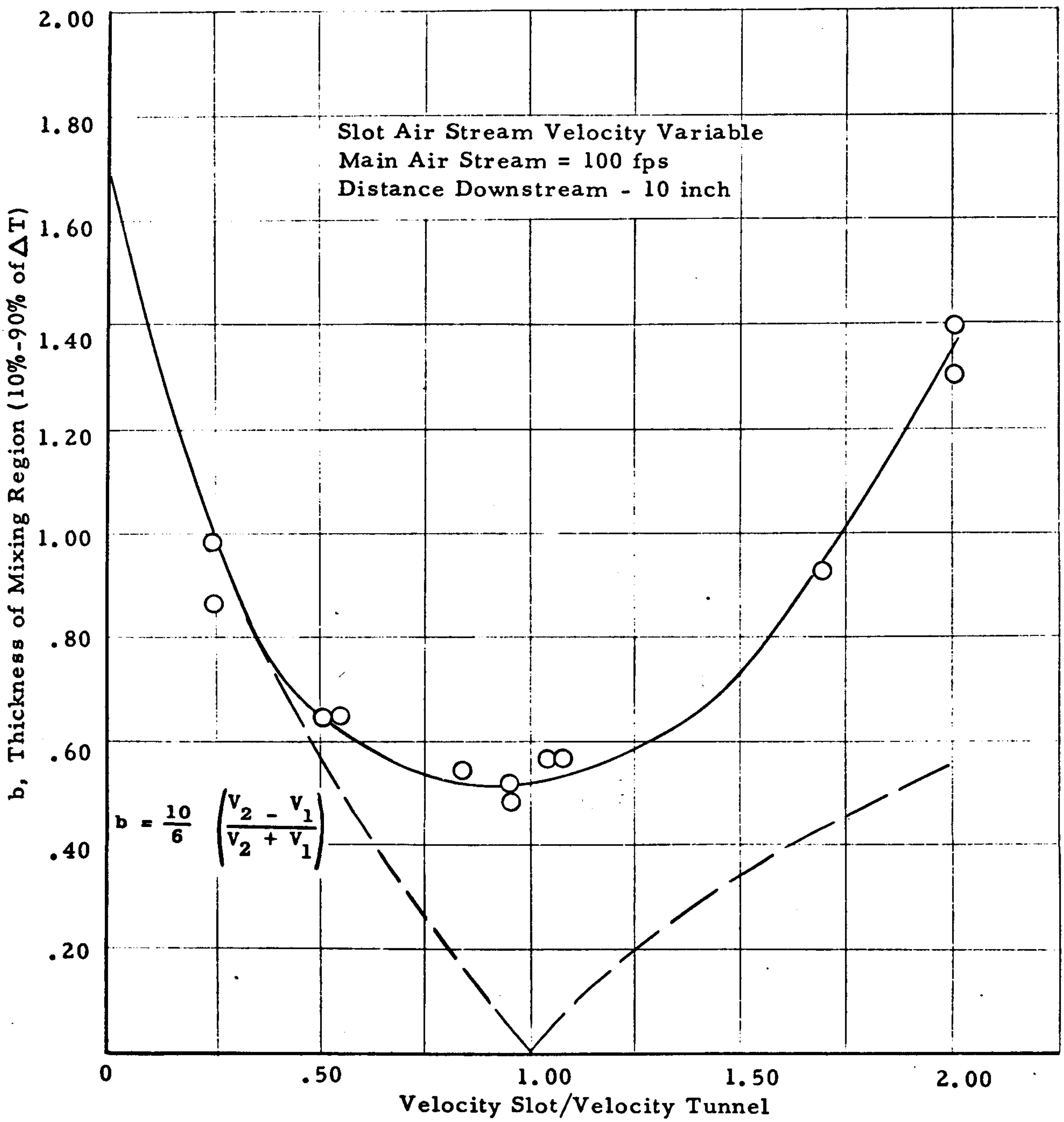


Figure 37: Effect of Approach Boundary Layer on Mixing Region

[redacted]

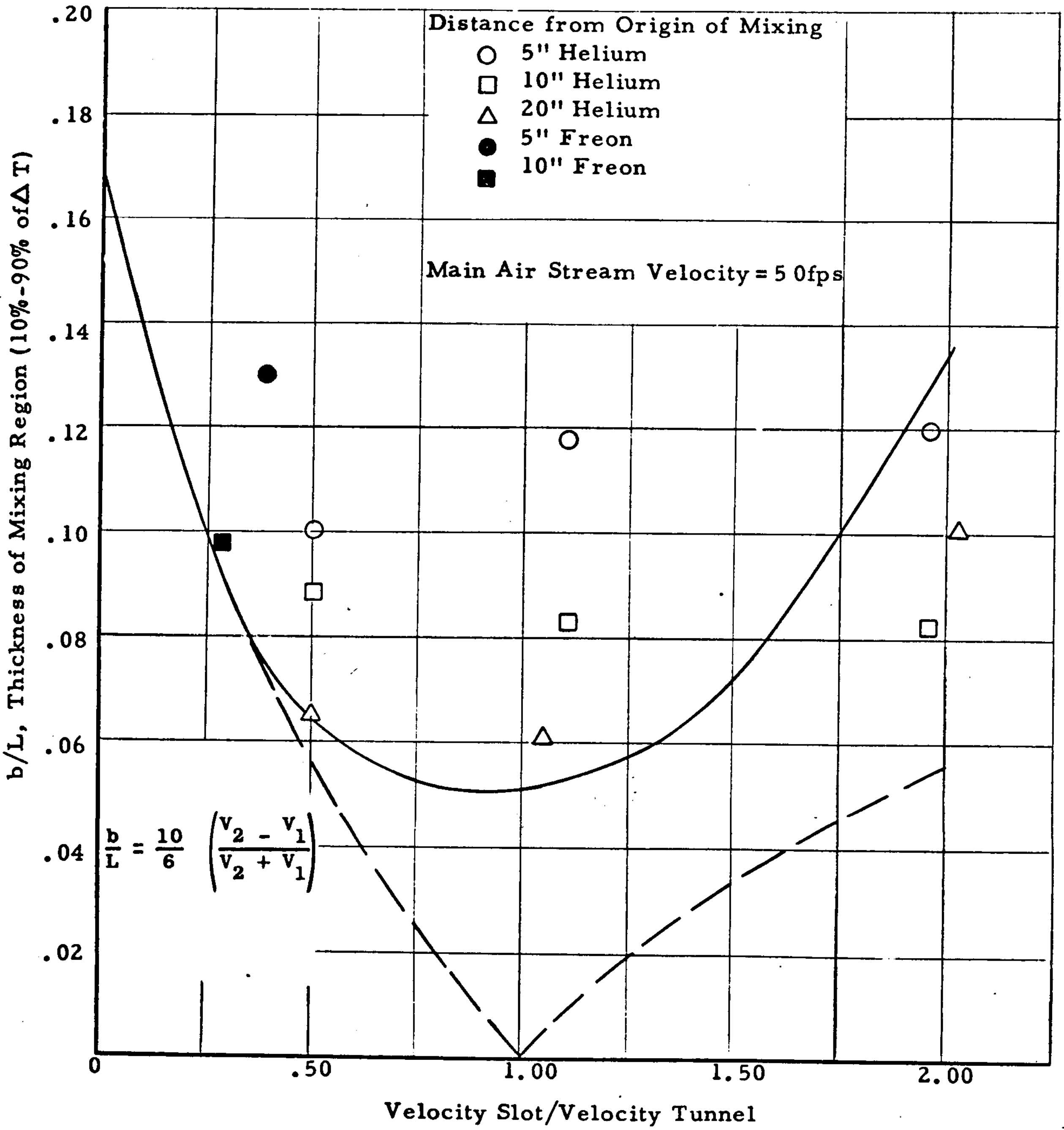


Figure 38: Effect of Approach Boundary Layer on Mixing Region

[redacted]

EFFICIENCY FROM MIXING THEORY

$\frac{1}{10} \times \frac{L}{D}$

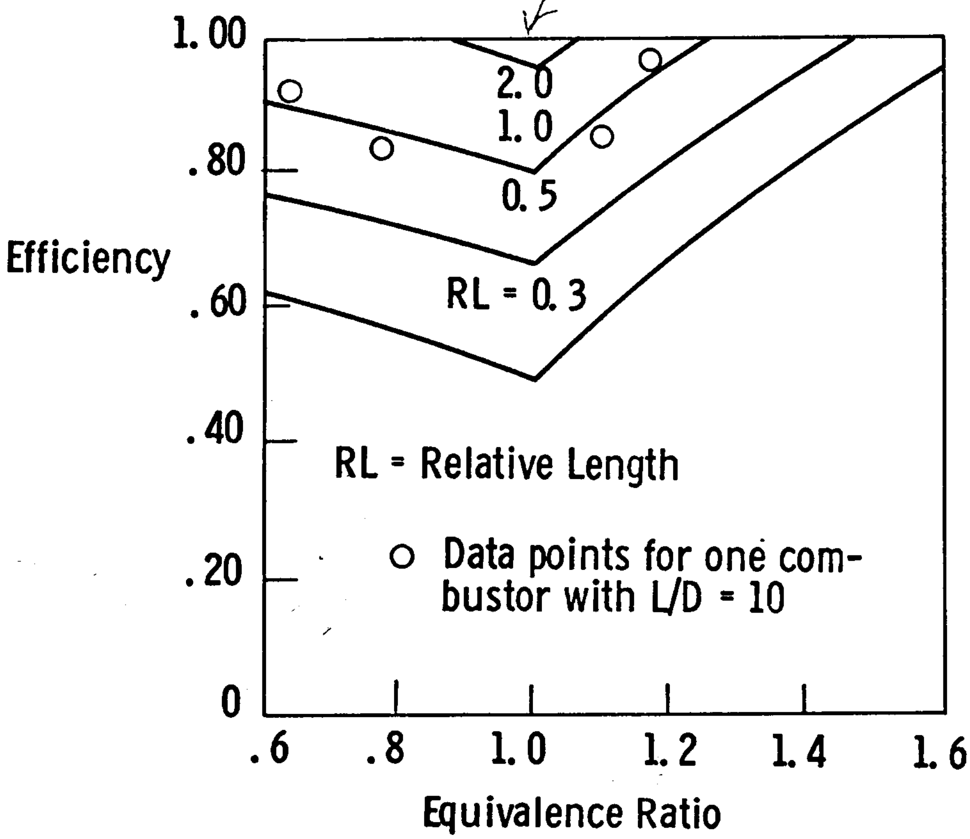
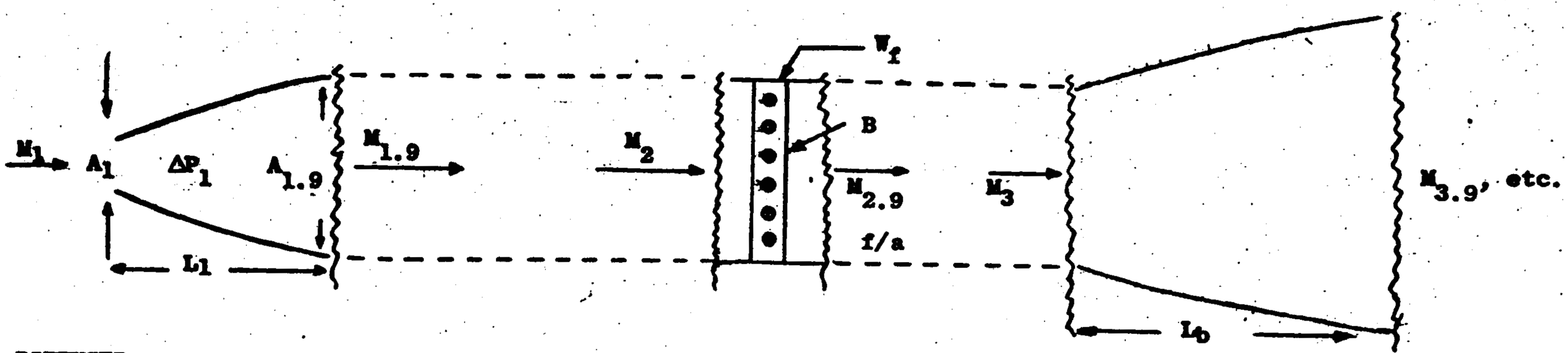


Figure 39:



DIFFUSER

GEOMETRIC VARIABLES

Inlet Area, A_1
Outlet Area, $A_{1.9}$
Diffuser Length, L_1

GAS-DYNAMIC VARIABLES

Inlet Mach No., M_1
(Pressure, Temperature, Weight Flow)

PERFORMANCE VARIABLES

Diffuser Dry Loss, ΔP_1
Outlet Mach No., $M_{1.9}$

FUEL INJECTOR

GEOMETRIC VARIABLES

$A_2 = A_{2.9}$
Blockage, B
Injector Size
Injector Spacing
Number of Orifices
Orifice Size

GAS-DYNAMIC VARIABLES

Air Inlet Mach No., M_2
(Pressure, Temperature, Weight Flow)
Fuel Flow Rate
(Temperature, Pressure)

PERFORMANCE VARIABLES

Fuel Source Pattern
Eddy Diffusivity, E
Fuel Penetration
Injector Dry Loss, ΔP_{2d}
Fuel Addition Momentum Loss, ΔP_{2m}
Outlet Mach No.
(Velocity, Temperature, Pressure, Fuel-Air Ratio)

COMBUSTOR

GEOMETRIC VARIABLES

Inlet Area, A_3
Outlet Area, $A_{3.9}$
Combustor Length (s)
(Variation of A vs. L)
Fuel Source Pattern

GAS-DYNAMIC VARIABLES

Eddy Diffusivity
Inlet Mach No., M_3
(Velocity, Temperature, Pressure)
Fuel-Air Ratio

PERFORMANCE VARIABLES
(as functions of L_b)

Outlet Mach No., $M_{3.9}$
Average Temperature, $T_{s3.9}$
Peak Temperature $T_{sP3.9}$
Static Pressure, $P_{s3.9}$
Combustion Efficiency, $N_{c3.9}$

Figure 40 - Overall Combustor Analysis Plan

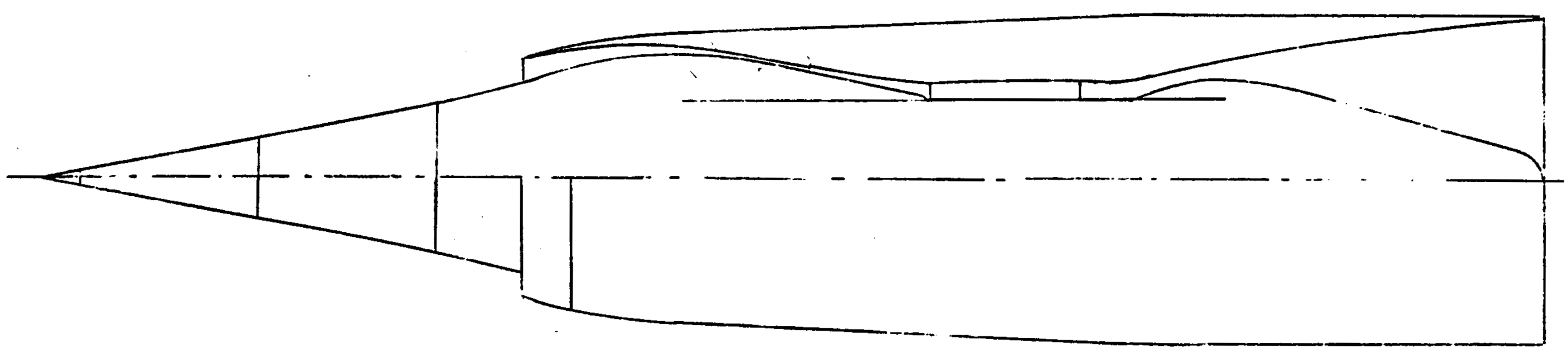


FIGURE 41: AXISYMMETRIC BURNER CONCEPT — VARIABLE INLET AND VARIABLE EXHAUST NOZZLE

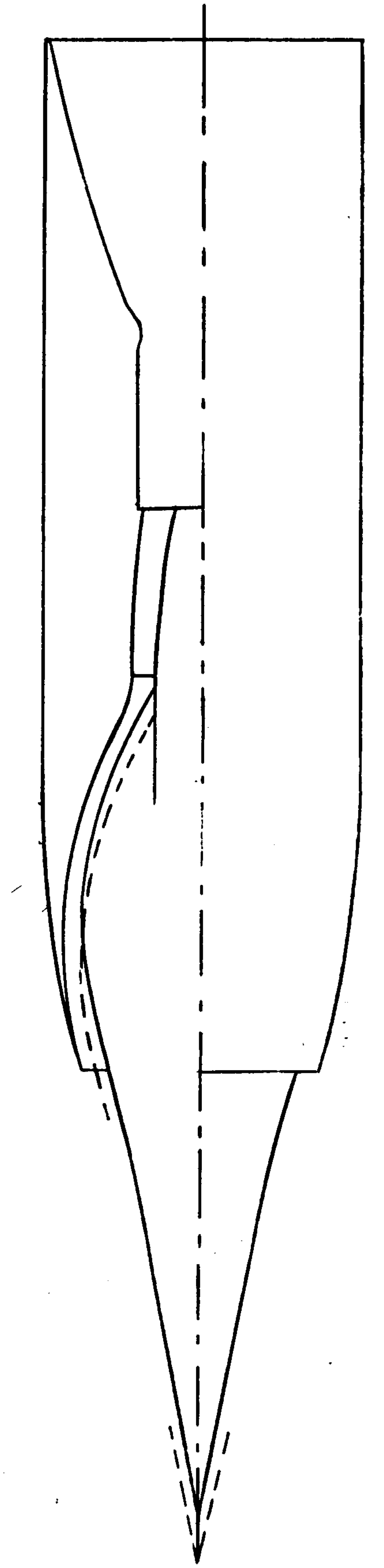
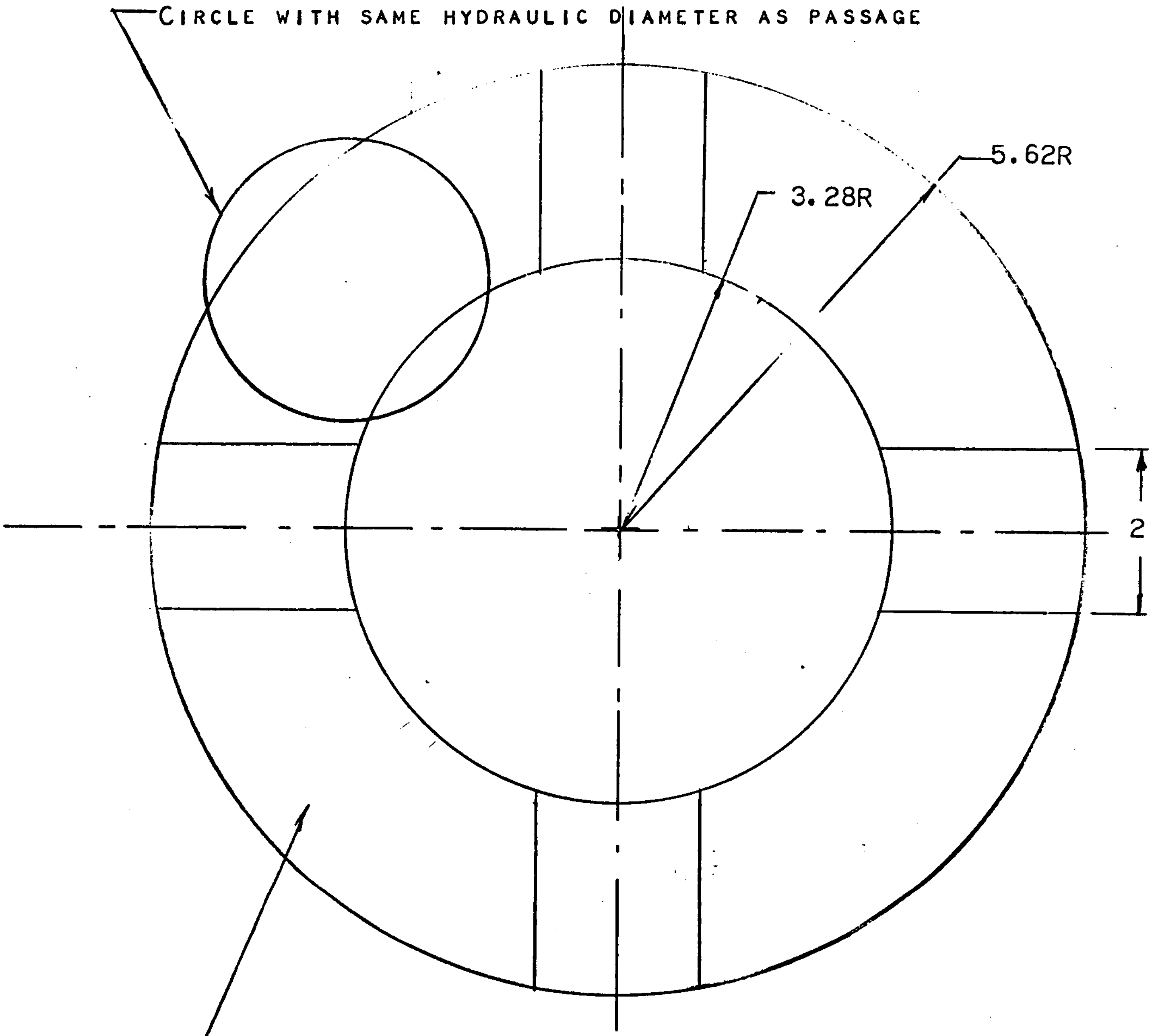


Figure 42: Axisymmetric Burner Concept — Annular Combustors, No Variable Exhaust Nozzle

Figure 42



$$\text{HYDRAULIC DIAMETER} = \frac{48/4}{\text{WALL CIRCUMFERENCE}} = 3.27''$$

$$\text{COMBUSTOR LENGTH} = \frac{L/D \times H_o}{\sqrt{\text{BURNER AREA RATIO}}} = 10 \times .632 \times 3.27 = 20.6''$$

FIGURE 43: COMPARISON OF CIRCULAR COMPONENT TEST BURNERS WITH ANNULAR COMBUSTOR PACKAGE

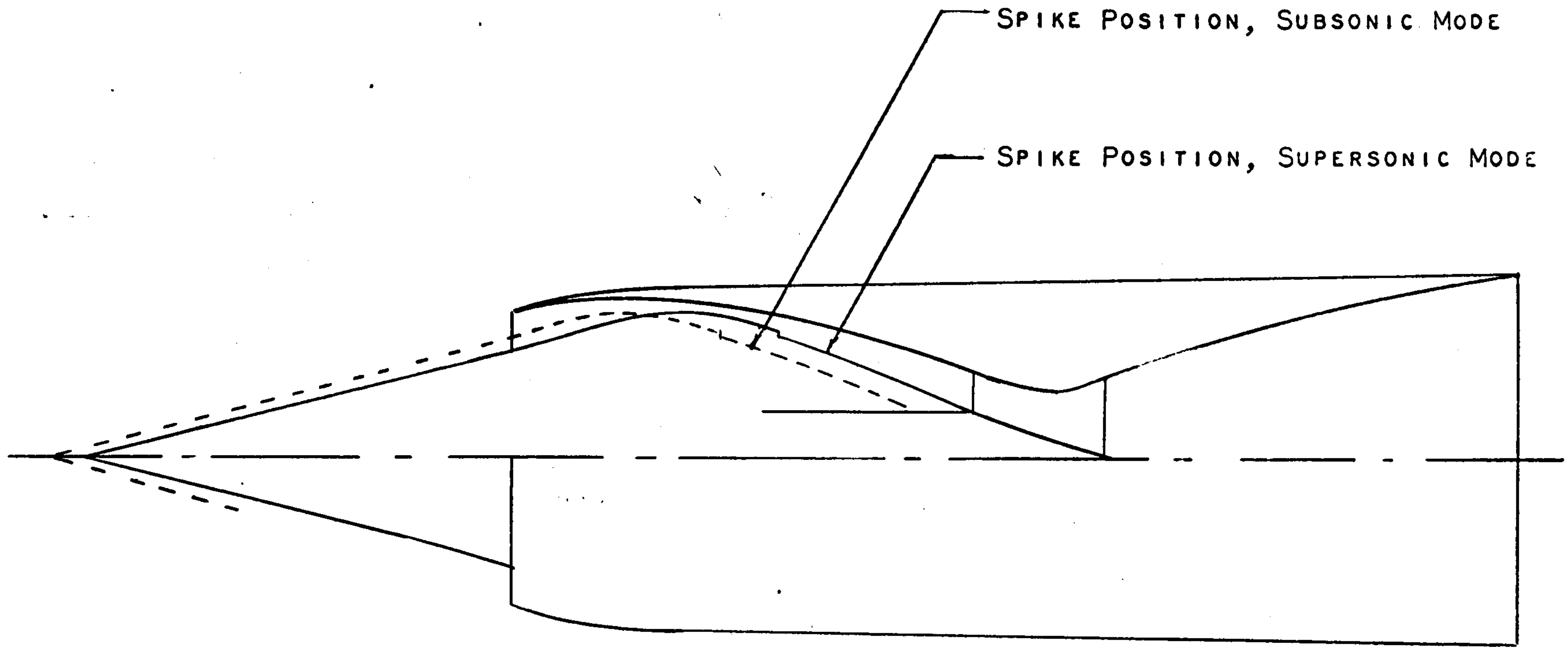


FIGURE 44: AXISYMMETRIC BURNER CONCEPT — COMBINED BURNERS

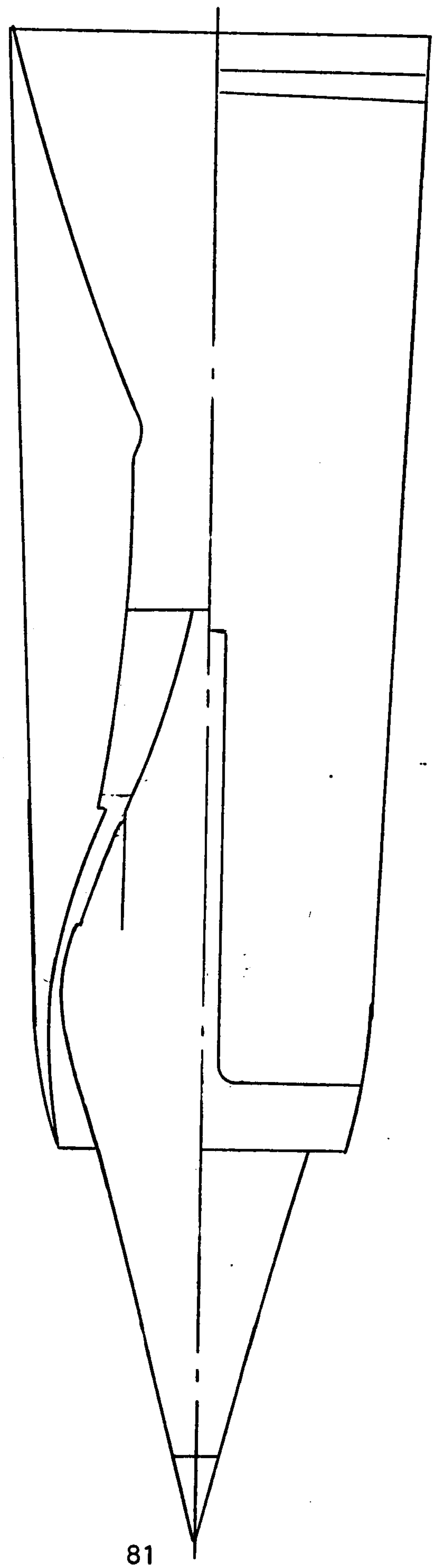


FIGURE 45: AXISYMMETRIC BURNER CONCEPT — BASELINE ENGINE

FIGURE 45

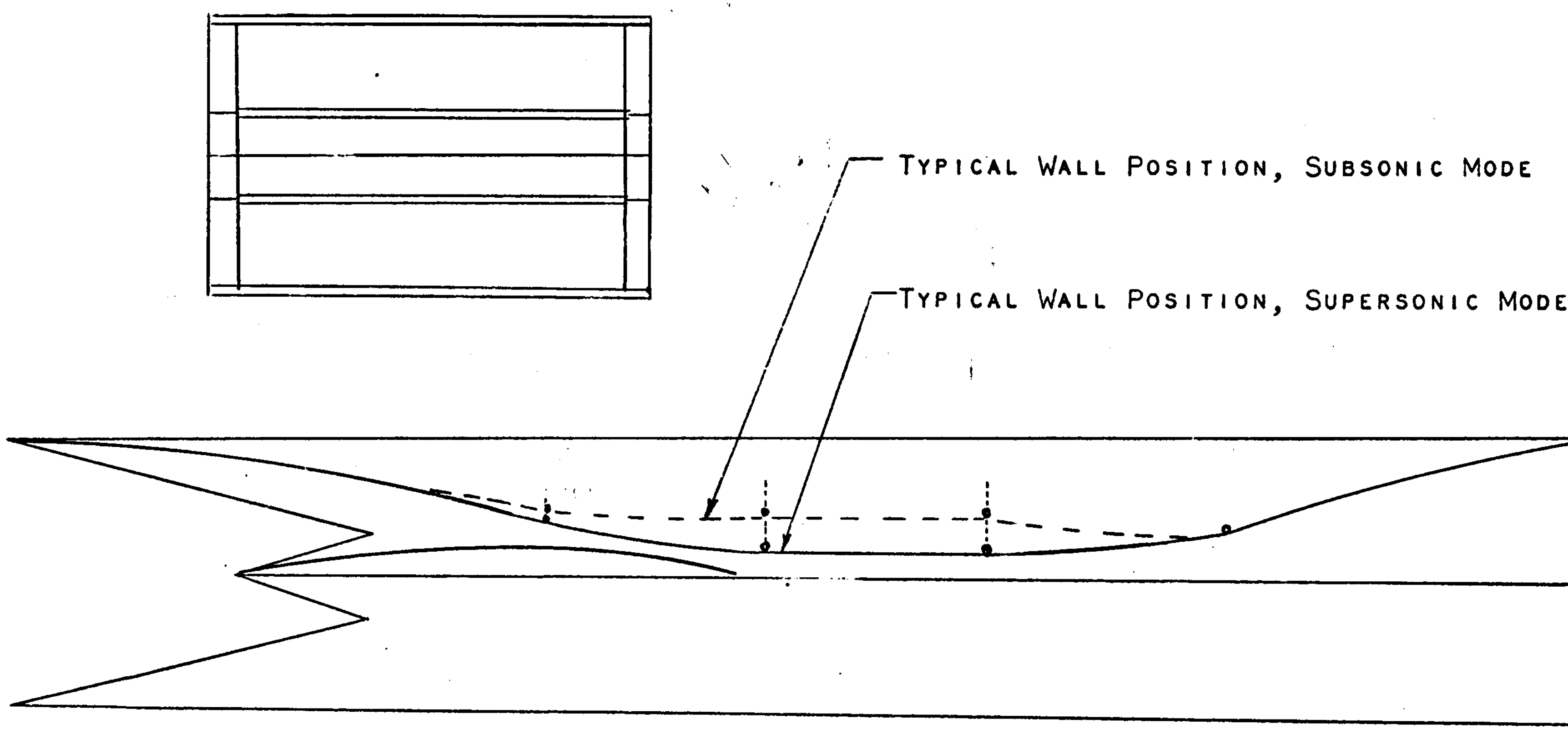


FIGURE 46: TWO DIMENSIONAL BURNER CONCEPT — HINGED WALLS

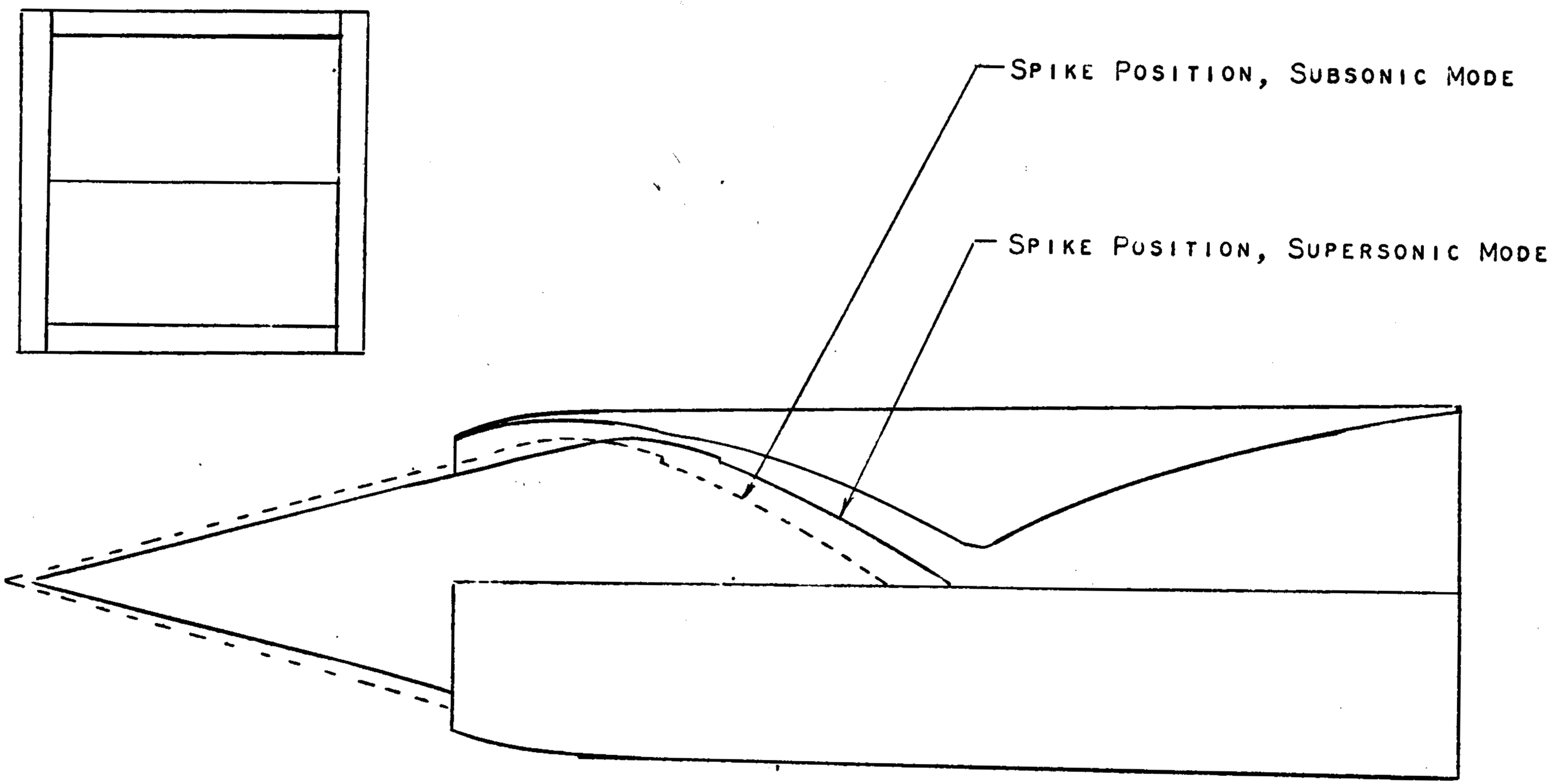


FIGURE 47: TWO-DIMENSIONAL BURNER CONCEPT — TRANSLATING SPIKE

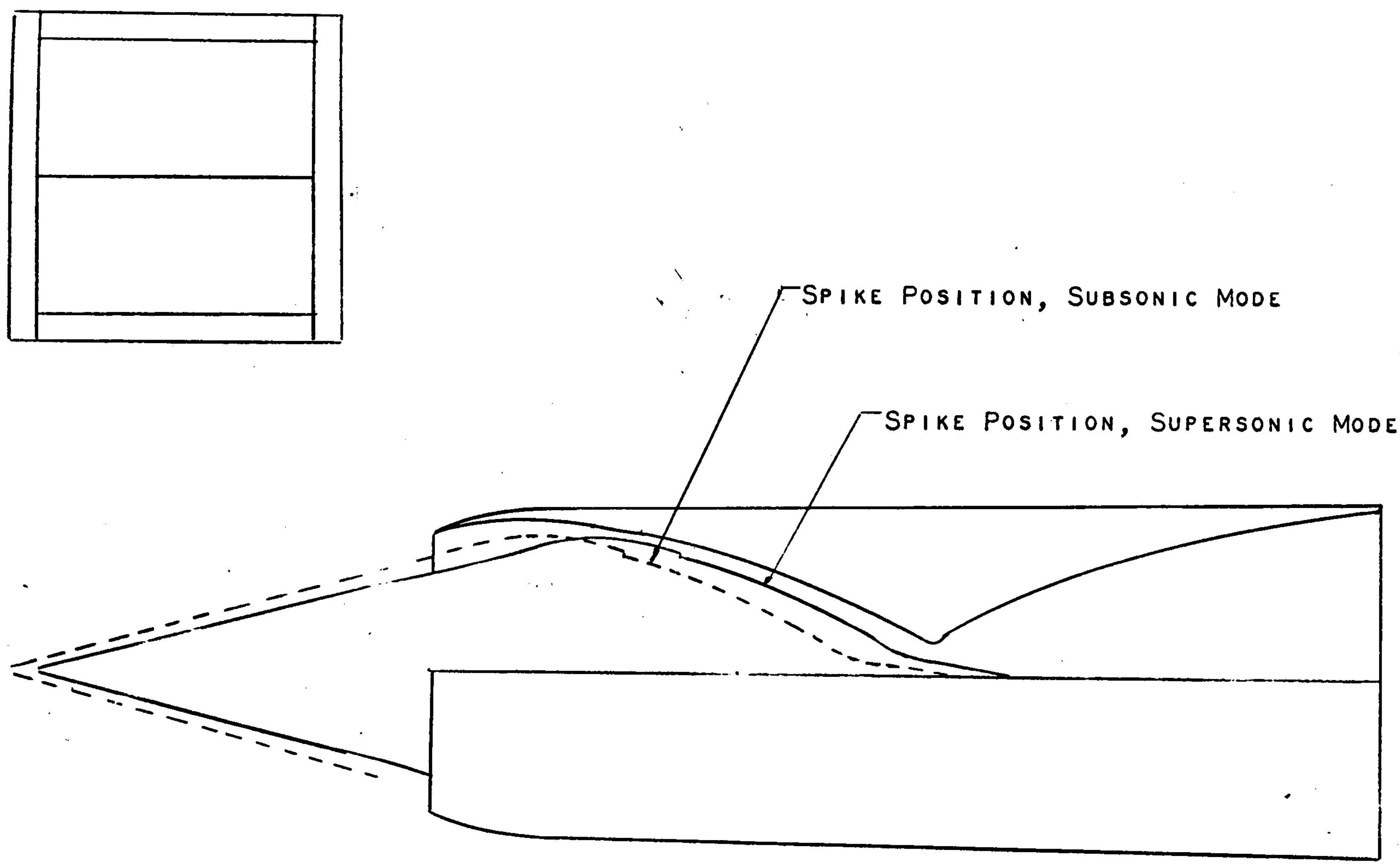


FIGURE 48: TWO-DIMENSIONAL COMBUSTOR DESIGN — TRANSLATING SPIKE WITH NOZZLE PLUG

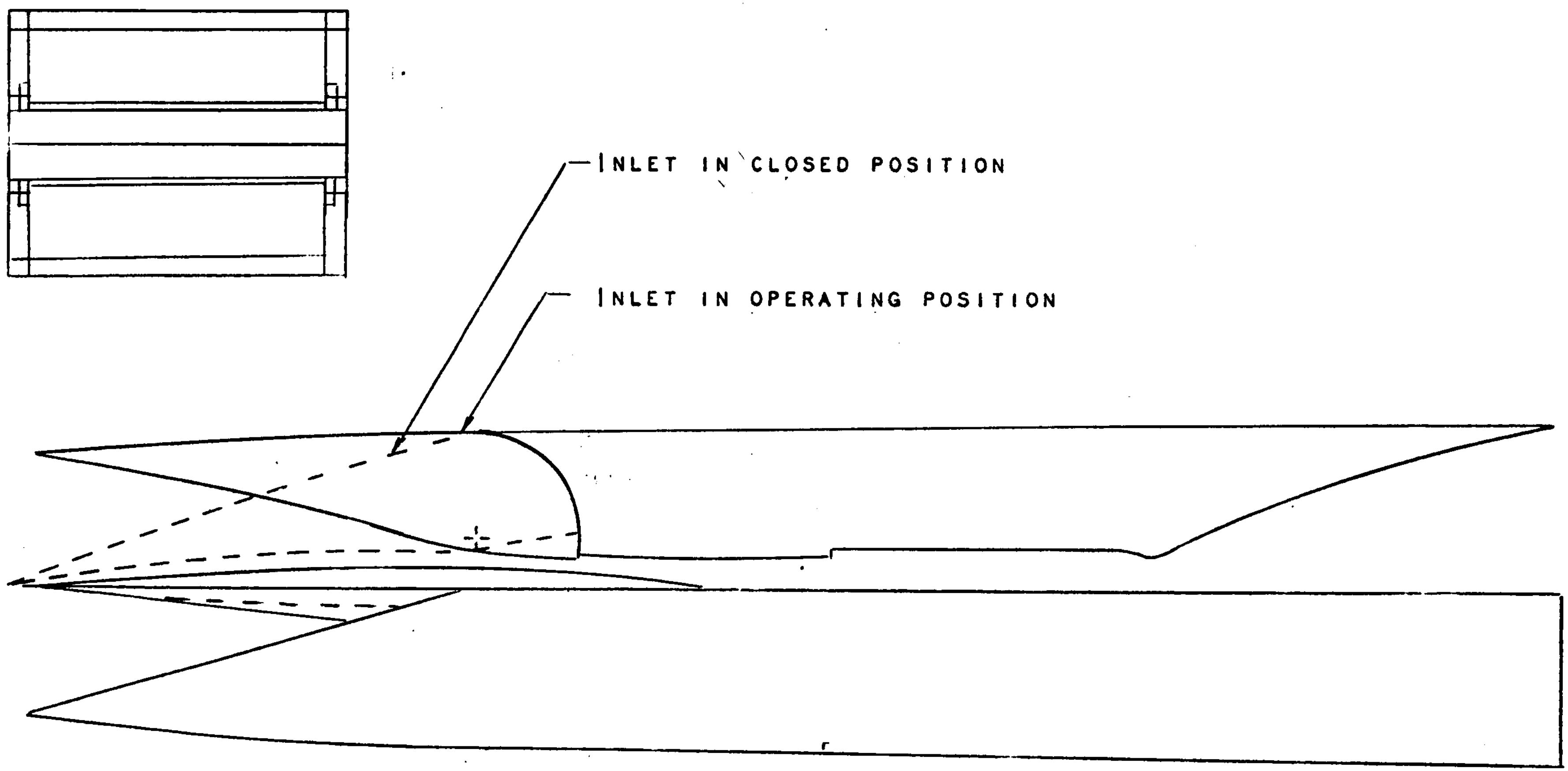


FIGURE 49: TWO-DIMENSIONAL BURNER CONCEPT — ROTATING COWL

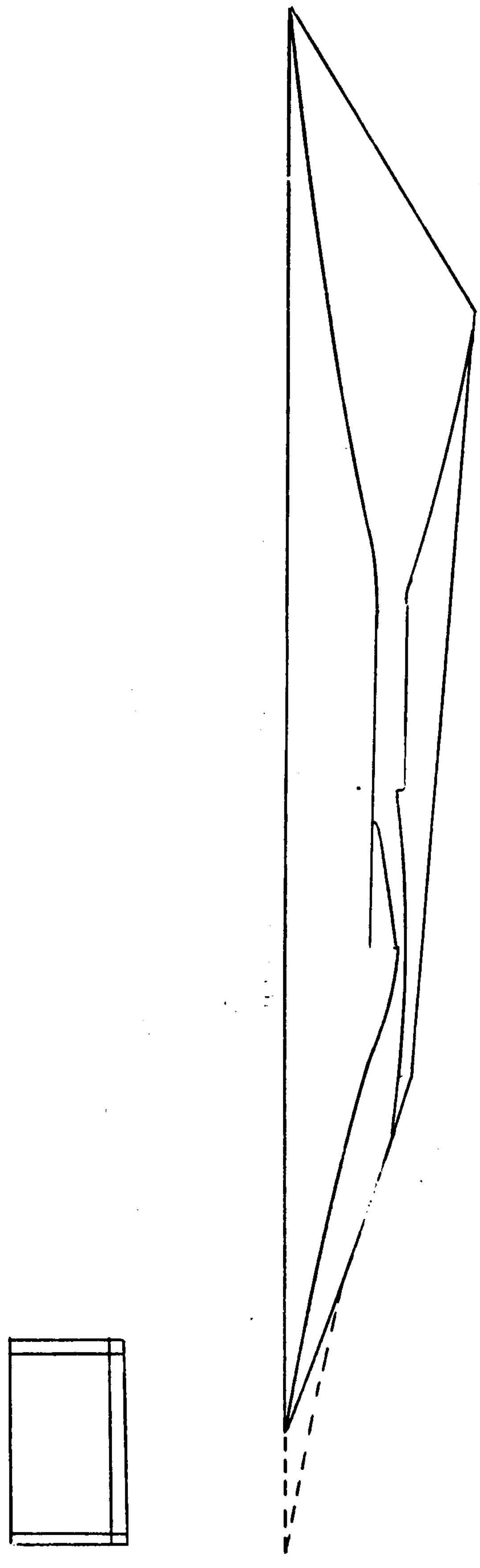


Figure 50: Two-Dimensional Asymmetric Burner Concept

Figure 50

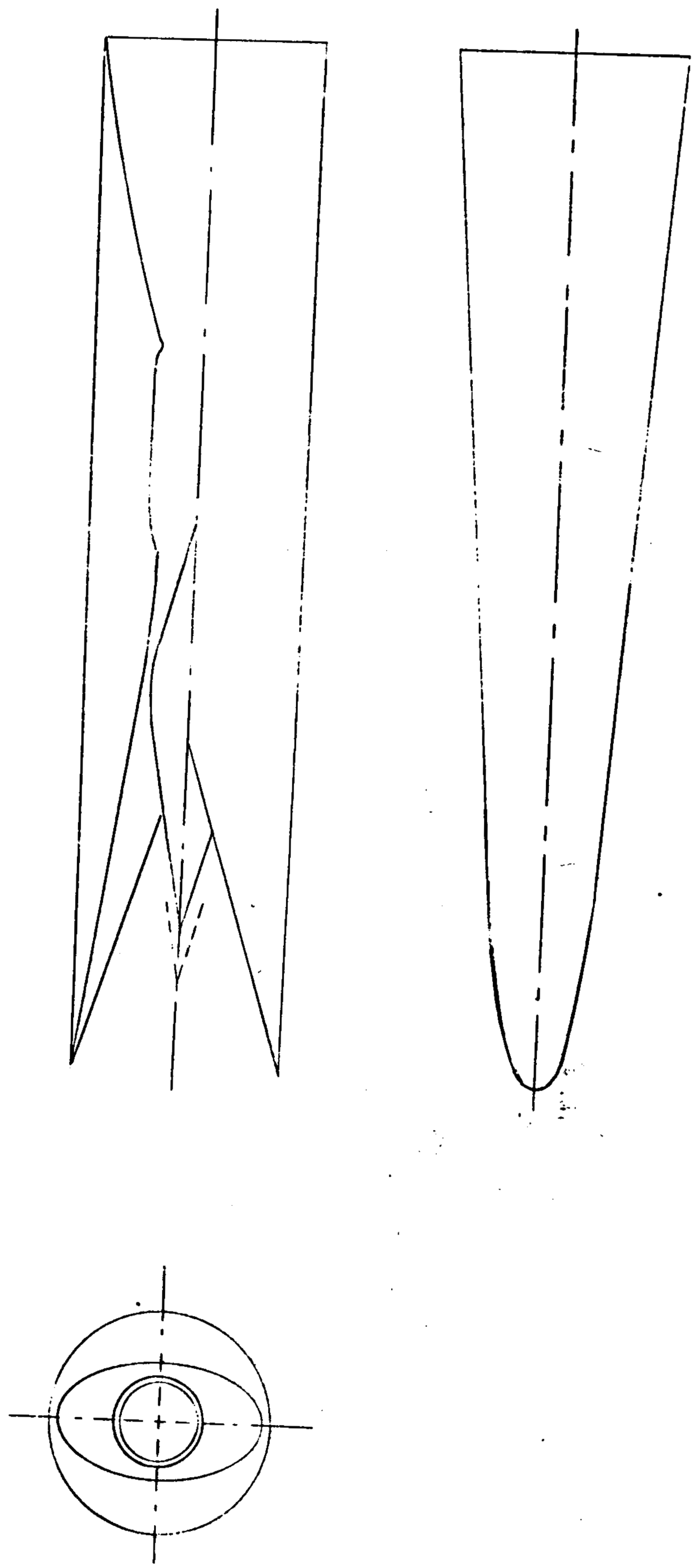
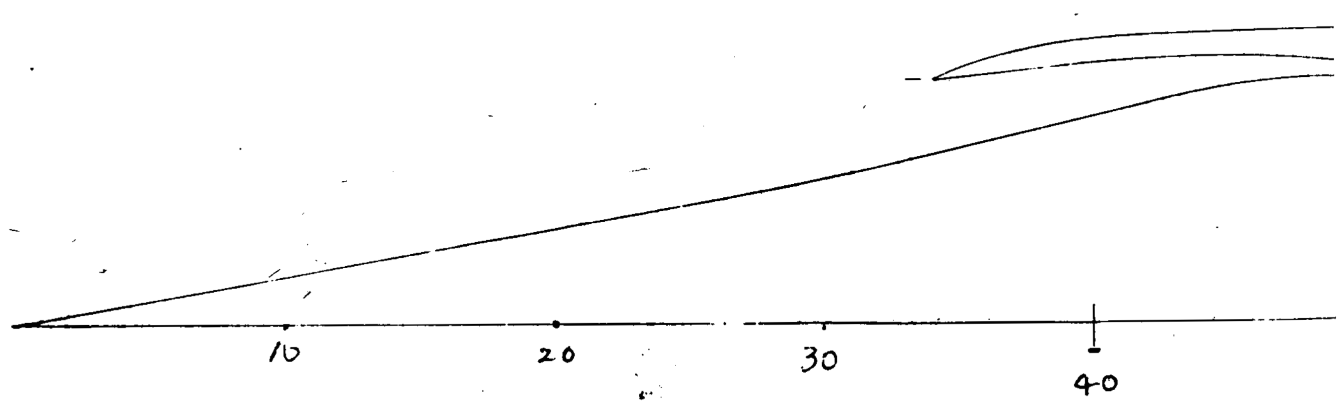


FIGURE 51: THREE-DIMENSIONAL COMBUSTOR CONCEPT

FIGURE 51


10
20
30
40

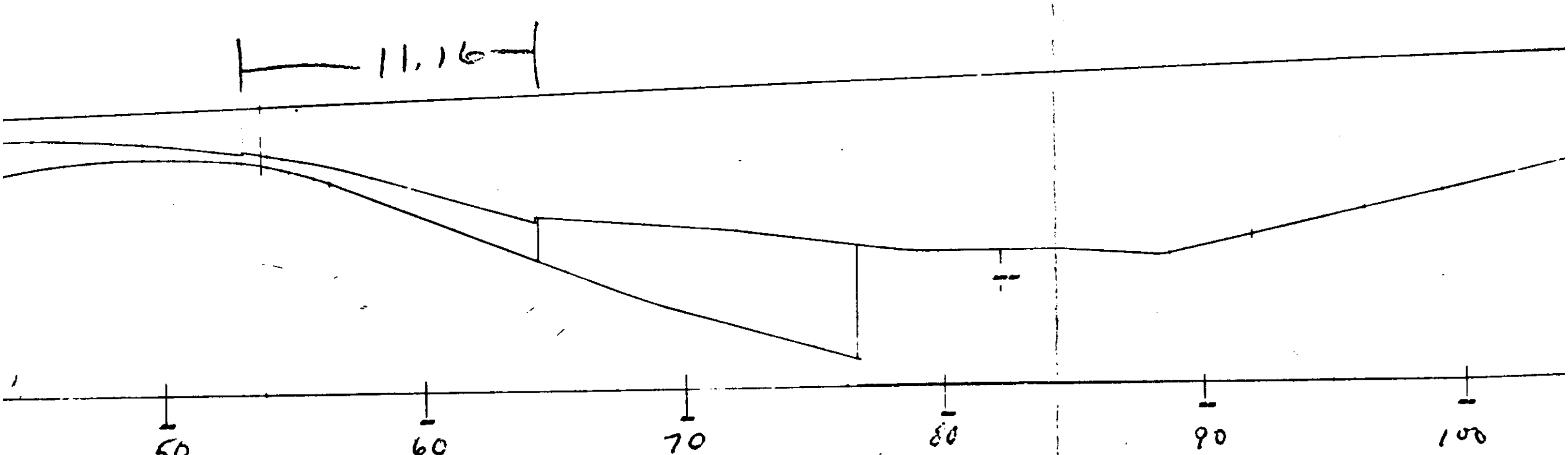


Figure 52: Engine Flow Path

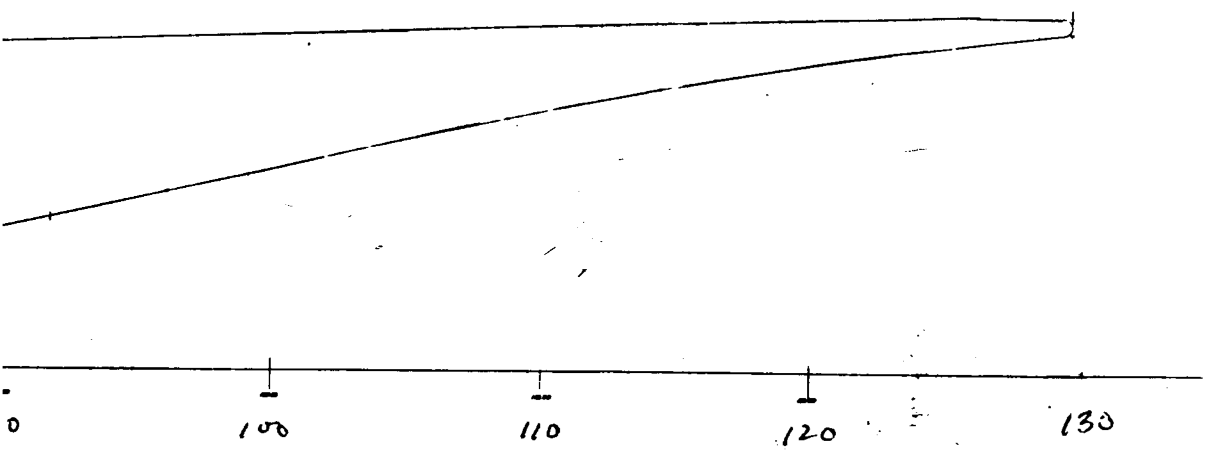
124

$2,5 \cdot 9$

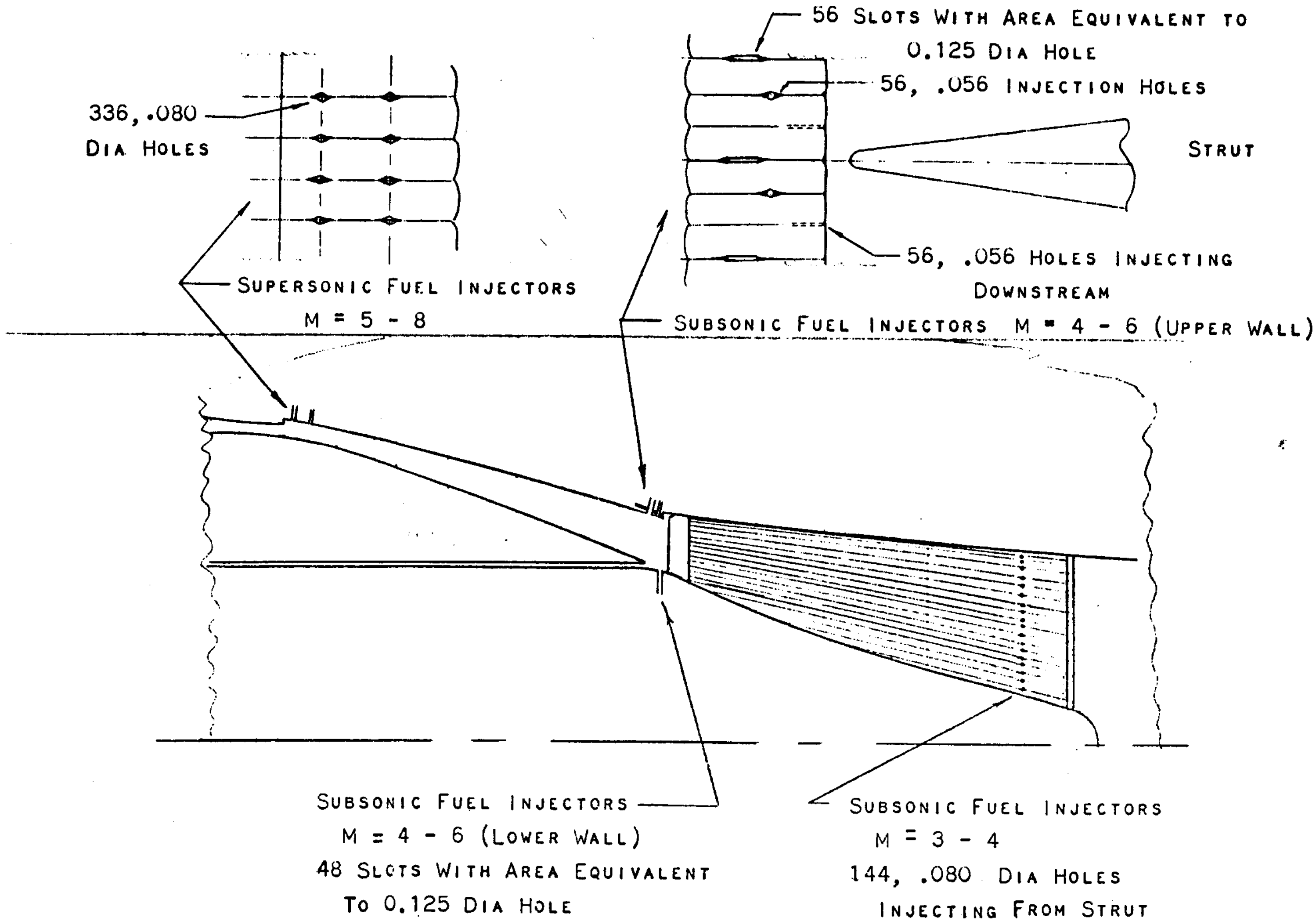
$$\frac{3,1}{2,5} \times 9$$

11,16

111



FUEL INJECTOR LOCATIONS - BASELINE ENGINE



336 .080
56 .125
56 .056
56 .056
48 .125
144 .080

696

3.97 □"

Figure 53:

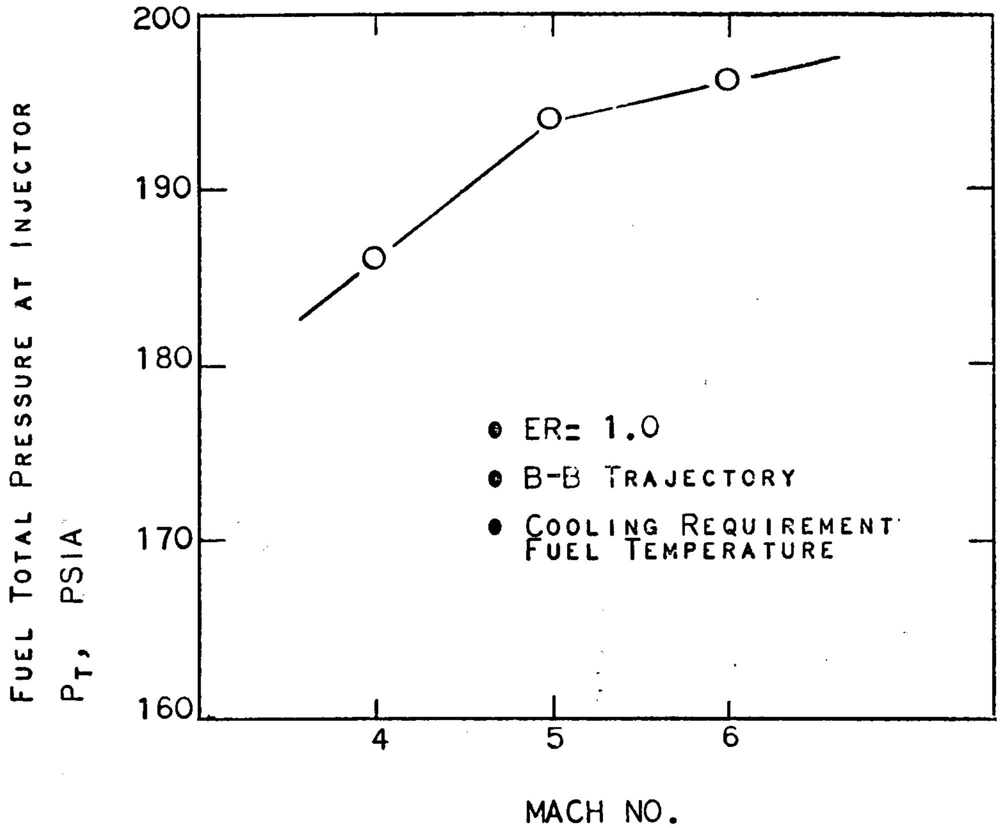


FIGURE 54: FUEL PRESSURE, SUBSONIC MODE

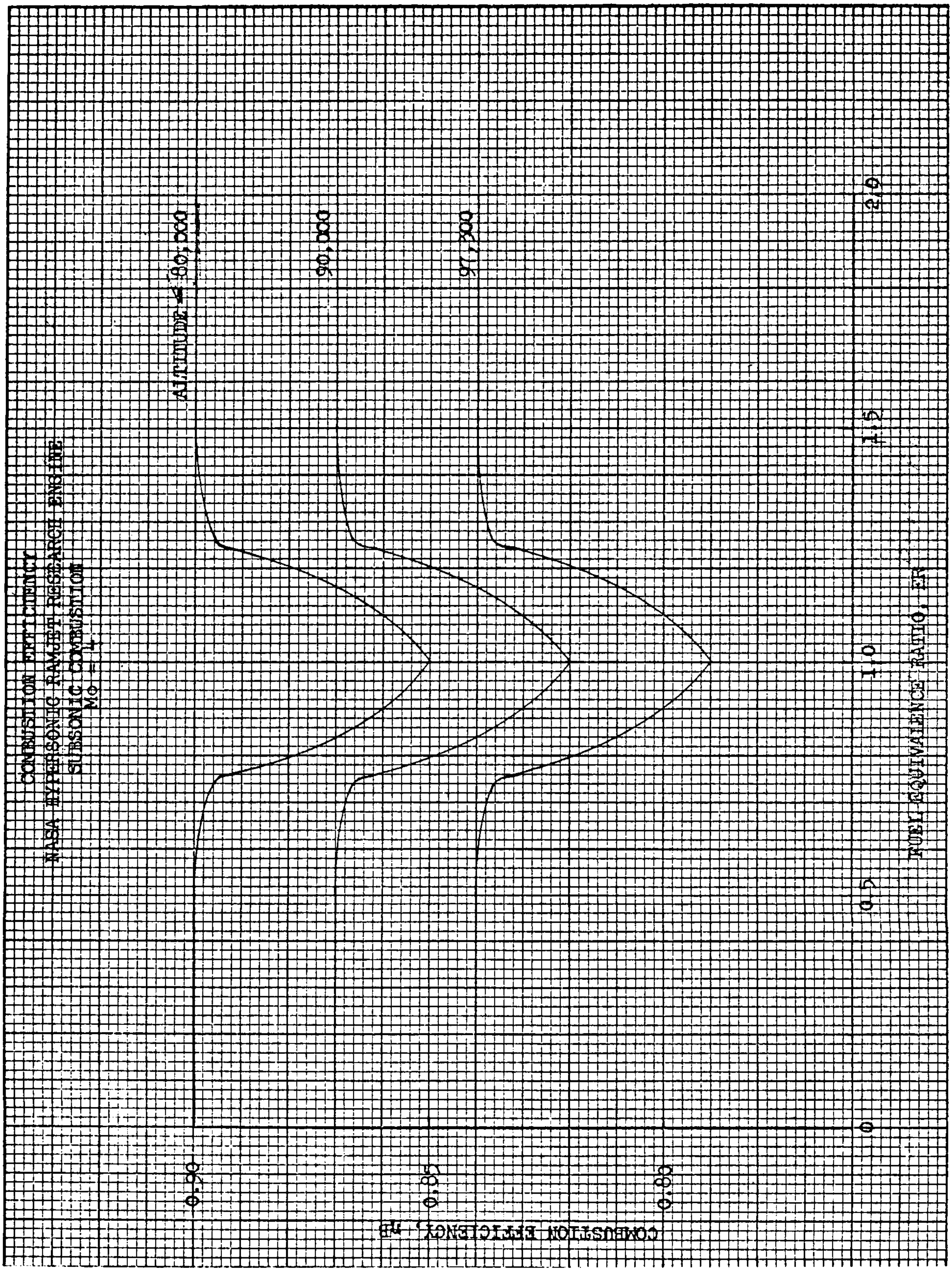

COMBUSTION EFFICIENCY
NASA HYPERSONIC RAMJET RESEARCH ENGINE
SUBSONIC COMBUSTION
Mo = 4
ALTITUDE = 80,000
90,000
97,300
0.90
0.85
0.80
COMBUSTION EFFICIENCY, ηB
0
0.5
1.0
1.5
2.0
FUEL EQUIVALENCE RATIO, ER

Figure 55

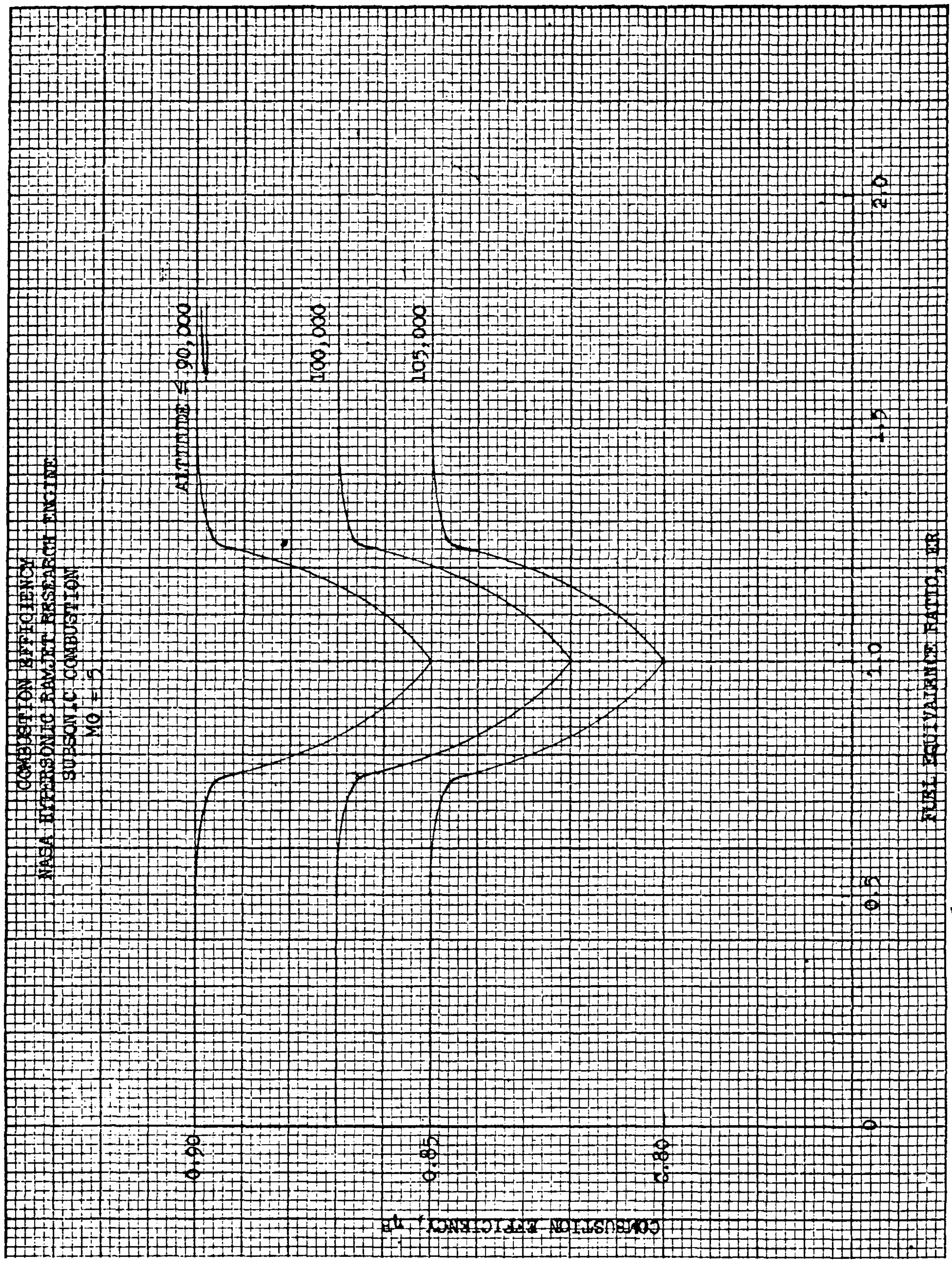


Fig 56

Figure 56

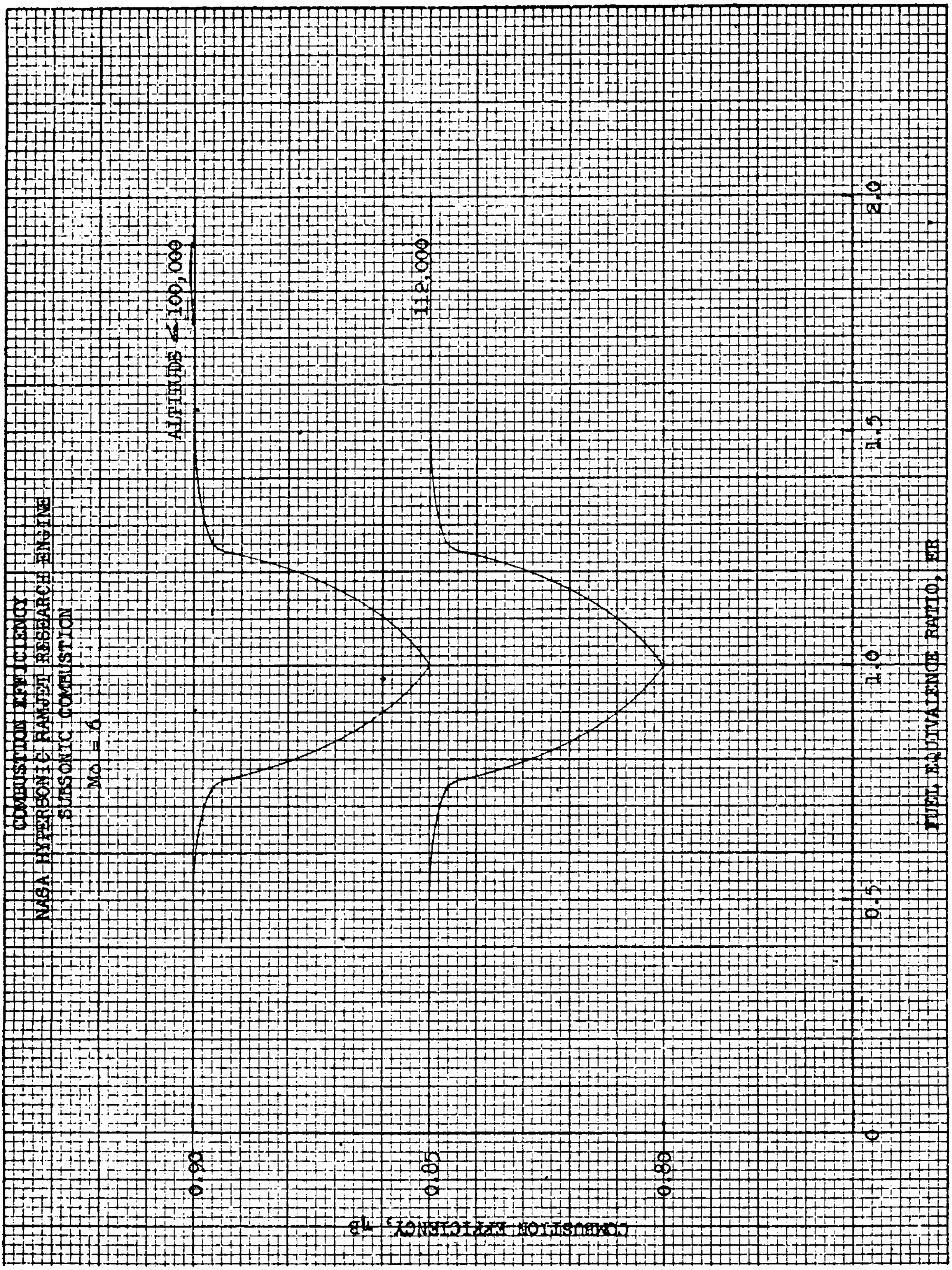

COMBUSTION EFFICIENCY
NASA HYPERSONIC RAMJET RESEARCH ENGINE
SUBSONIC COMBUSTION
Mo = 6
ALTITUDE ≤ 100,000
112,000
0.90
0.85
0.80
0
0.5
1.0
1.5
2.0
FUEL EQUIVALENCE RATIO, ER
COMBUSTION EFFICIENCY, ηB

Figure 57

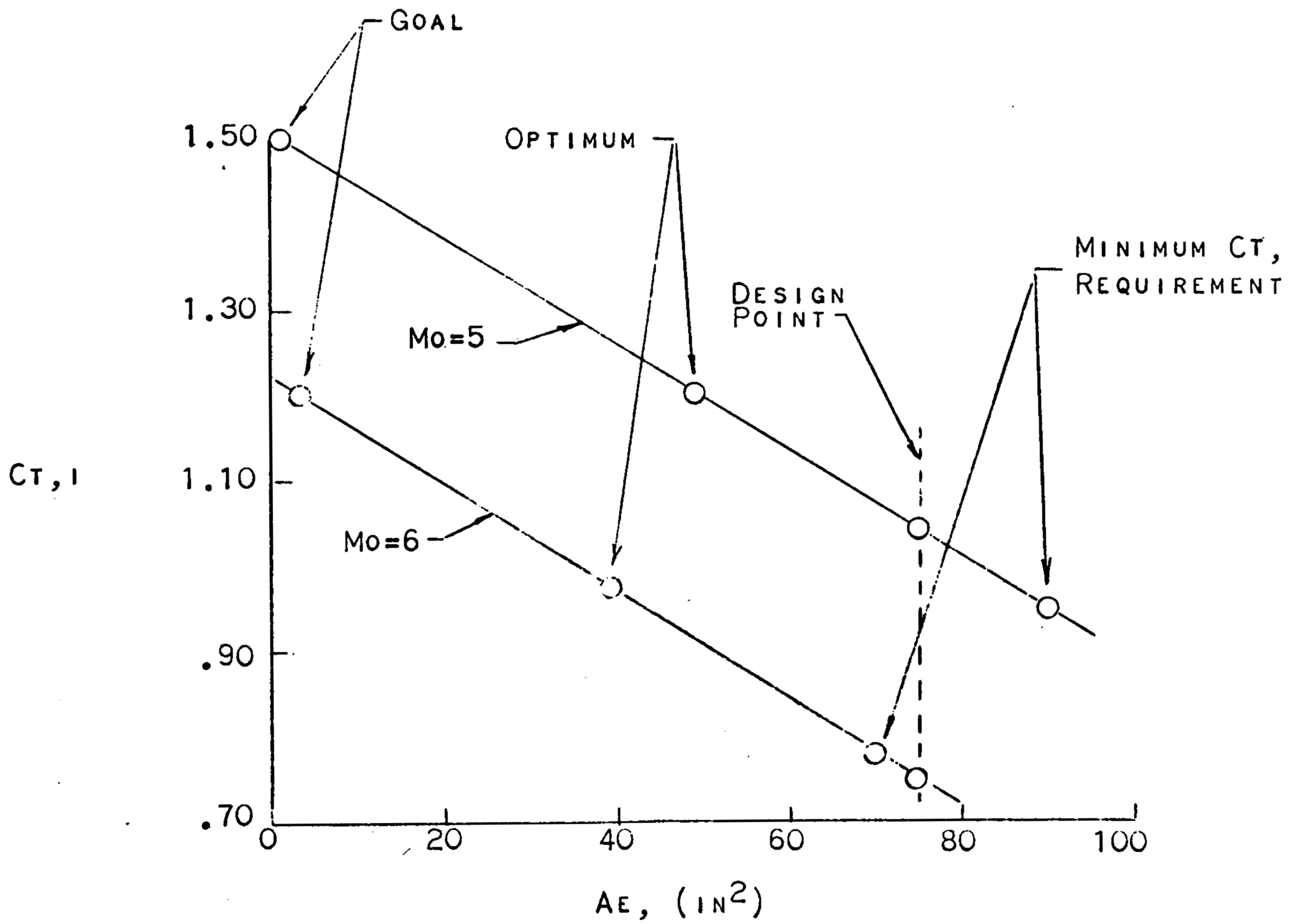


FIGURE 58: ENGINE THRUST COEFFICIENT VERSUS EXHAUST NOZZLE THRUST AREA

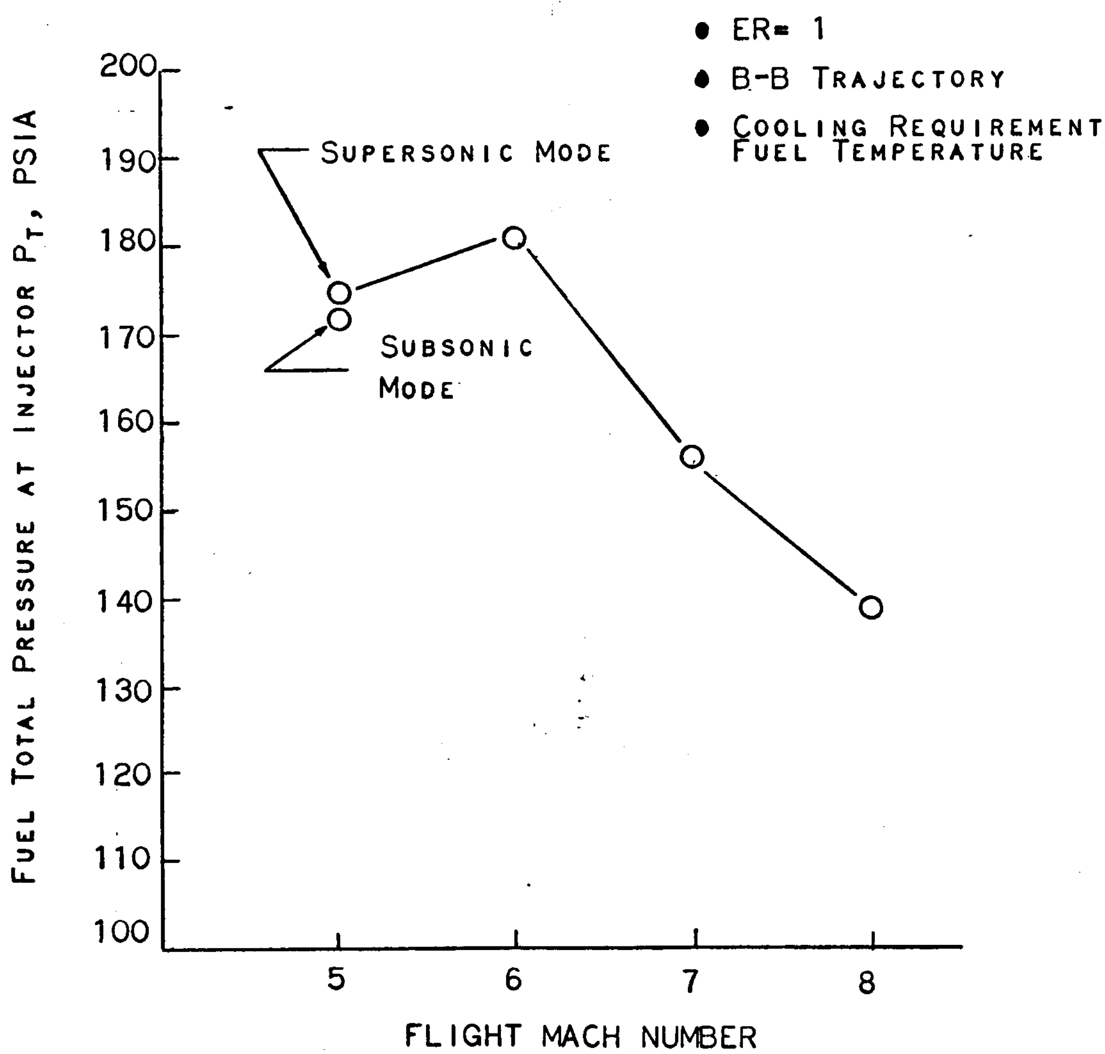


FIGURE 59: FUEL MANIFOLD PRESSURE, SUPERSONIC MODE

CALCULATED PENETRATION AS A FUNCTION OF

FLIGHT MACH NUMBER

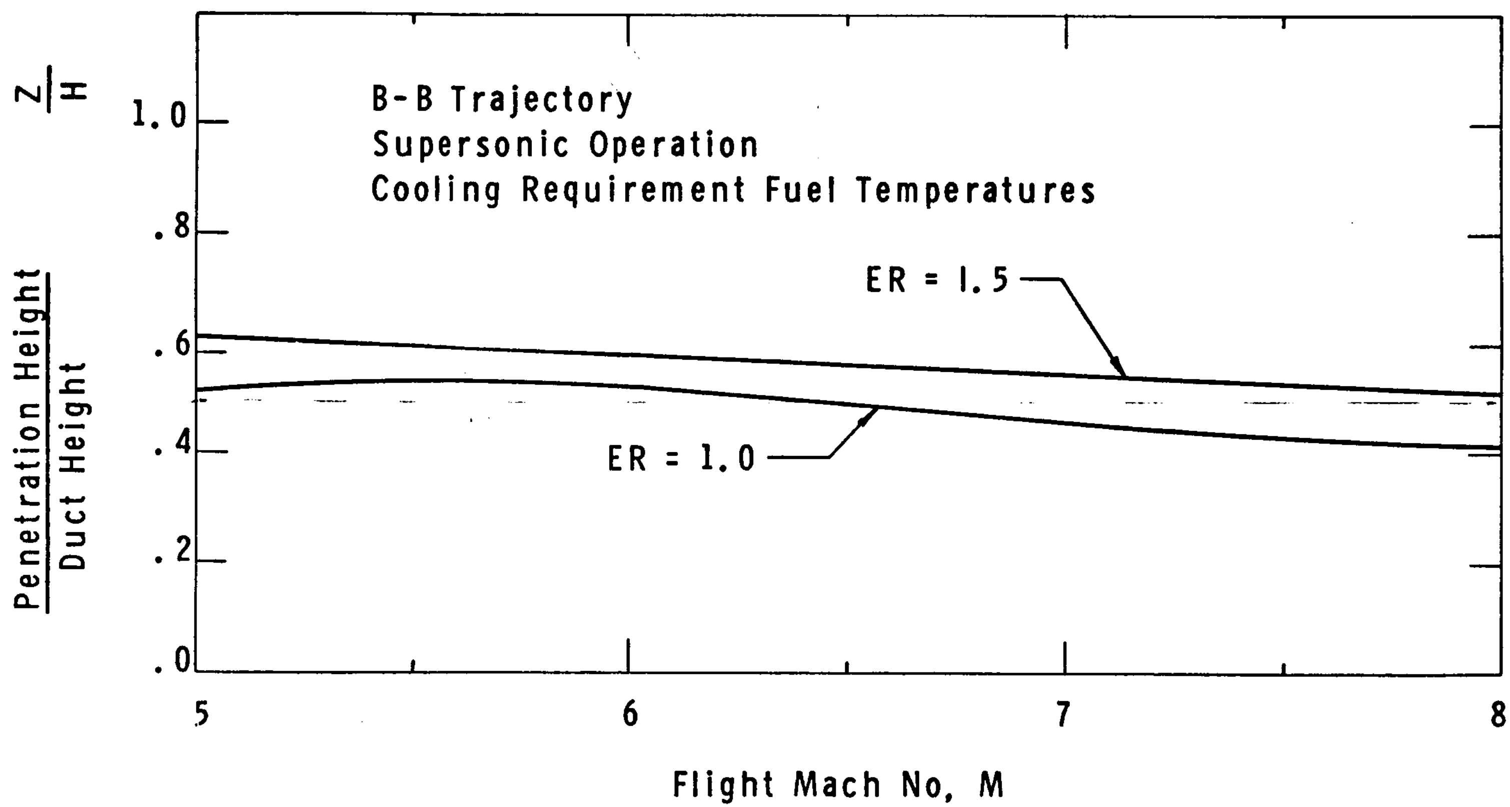
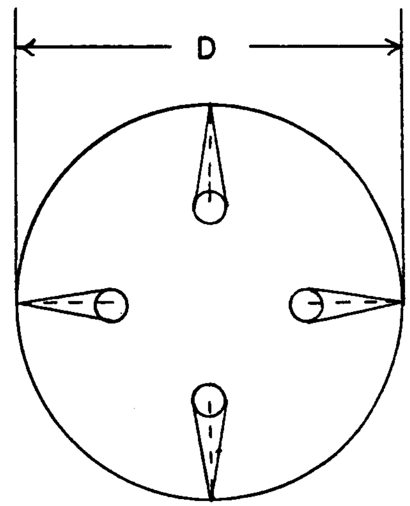
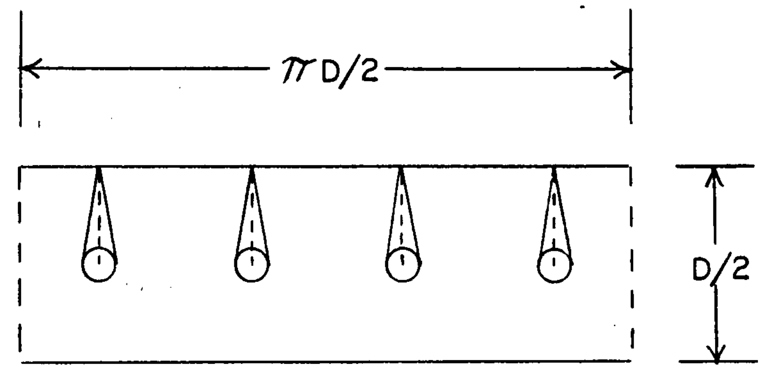


Figure 60:



4 INJECTORS
PENETRATION = D/4
AREA = $\pi D^2/4$
DISTANCE BETWEEN VIRTUAL SOURCES = .353 D



4 INJECTORS
PENETRATION = D/4
AREA = $\pi D^2/4$
DISTANCE BETWEEN VIRTUAL SOURCES = .392D

FIGURE 61: CIRCULAR-ANNULAR INJECTION COMPARISON

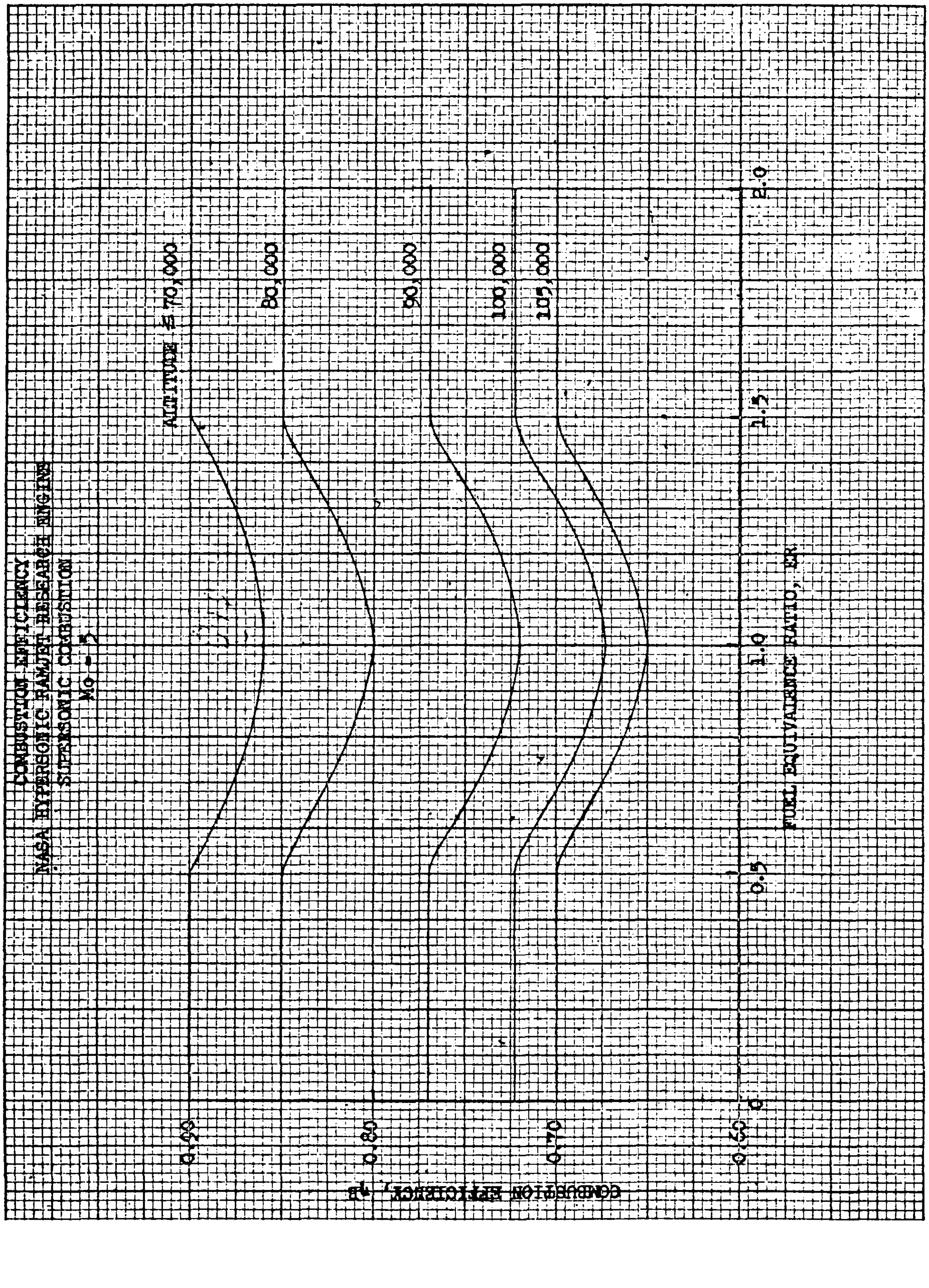

COMBUSTION EFFICIENCY
NASA HYPERSONIC RAMJET RESEARCH ENGINE
SUPERSONIC COMBUSTION
Mo = 5
ALTITUDE ≦ 70,000
80,000
90,000
100,000
105,000
0.90
0.80
0.70
0.60
COMBUSTION EFFICIENCY, ηB
0
0.5
1.0
1.5
2.0
FUEL EQUIVALENCE RATIO, ER

Figure 62

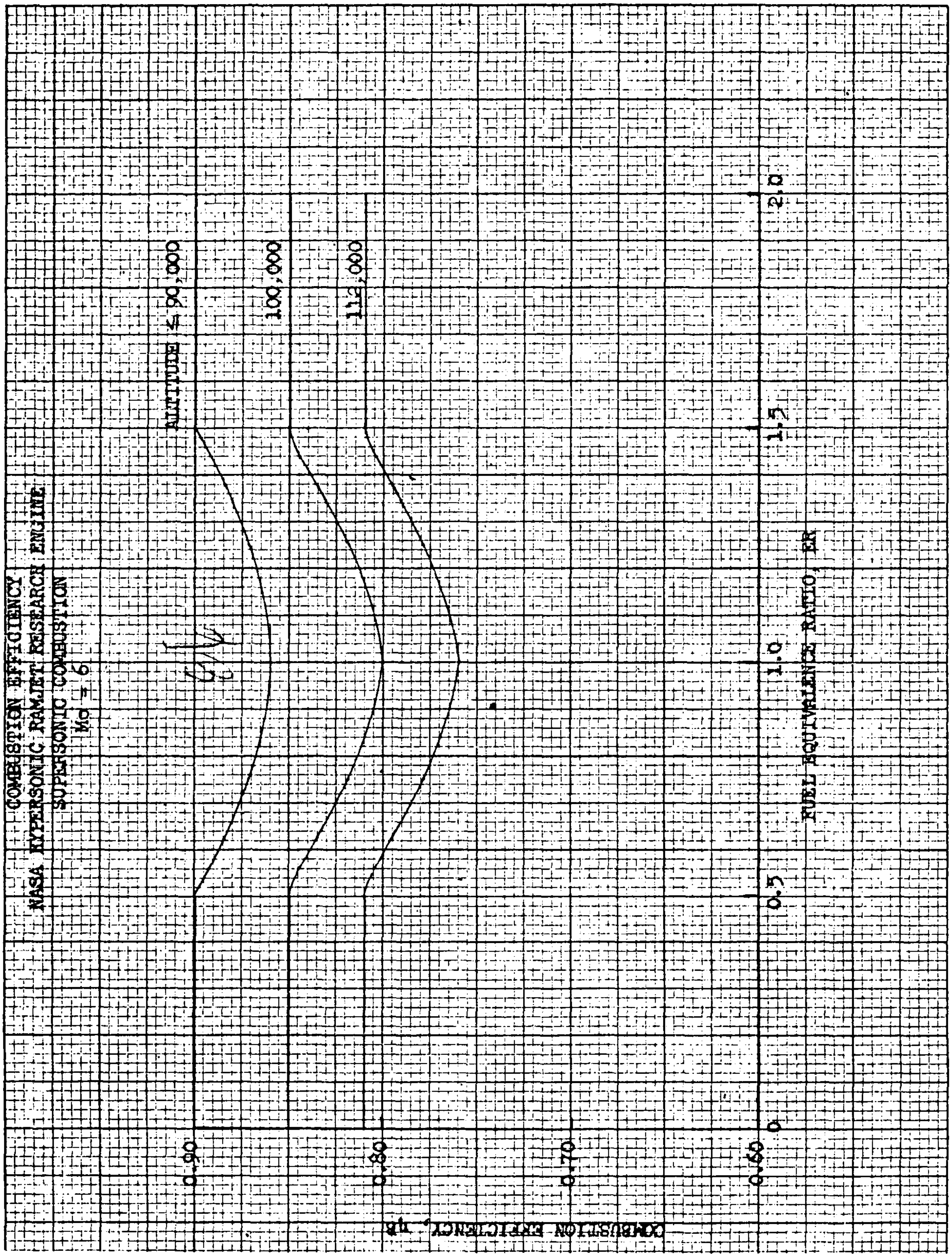

COMBUSTION EFFICIENCY
NASA HYPERSONIC RAMJET RESEARCH ENGINE
SUPERSONIC COMBUSTION
Mo = 6
ALTITUDE ≤ 90,000
100,000
112,000
0.90
0.80
0.70
0.60
0
0.5
1.0
1.5
2.0
FUEL EQUIVALENCE RATIO, ER
COMBUSTION EFFICIENCY, ηB

Figure 63

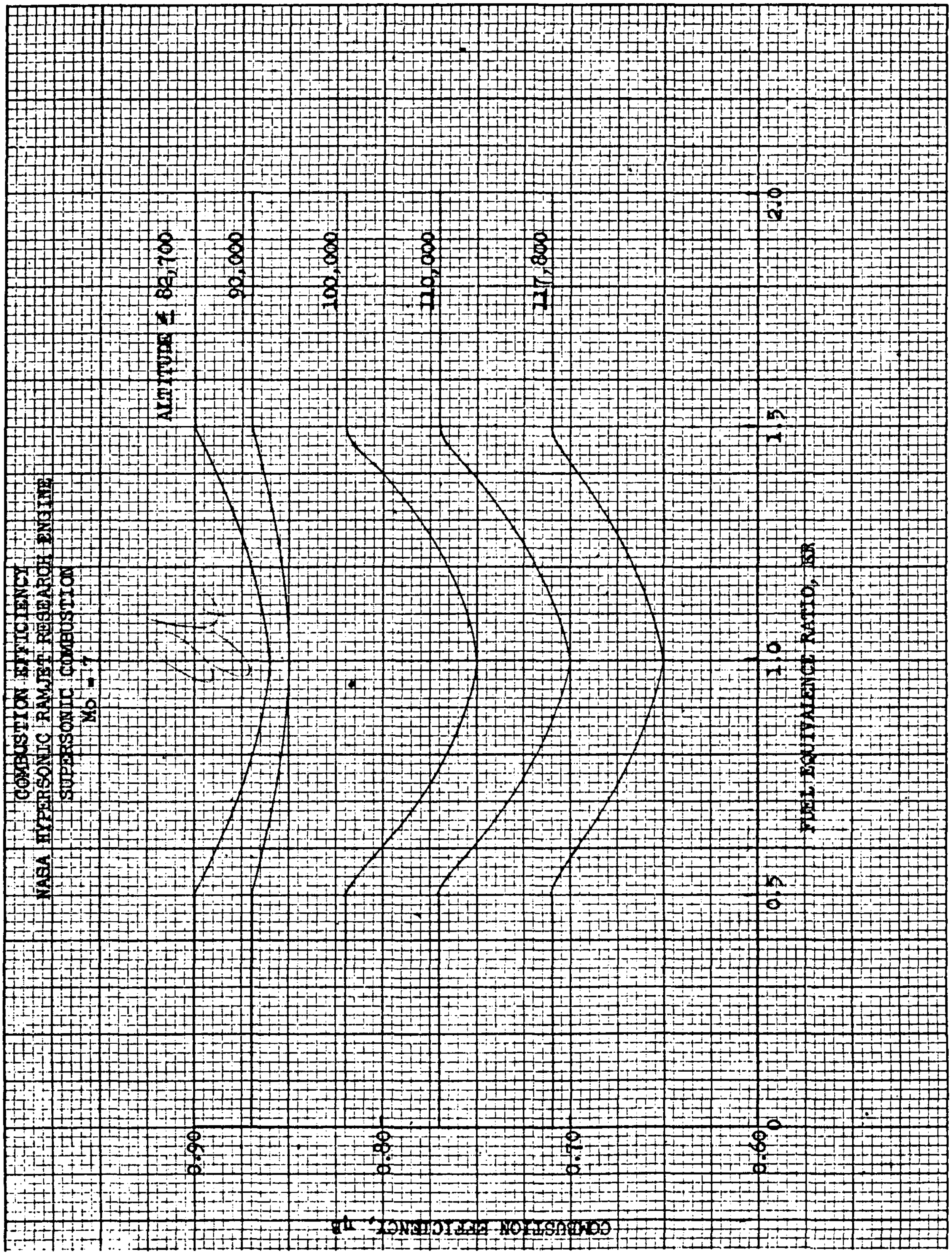

COMBUSTION EFFICIENCY
NASA HYPERSONIC RAMJET RESEARCH ENGINE
SUPERSONIC COMBUSTION
M0 = 7
ALTITUDE = 82,700
90,000
100,000
110,000
117,800
0.90
0.80
0.70
0.60
0
0.5
1.0
1.5
2.0
FUEL EQUIVALENCE RATIO, ER
COMBUSTION EFFICIENCY, ηB

Figure 64

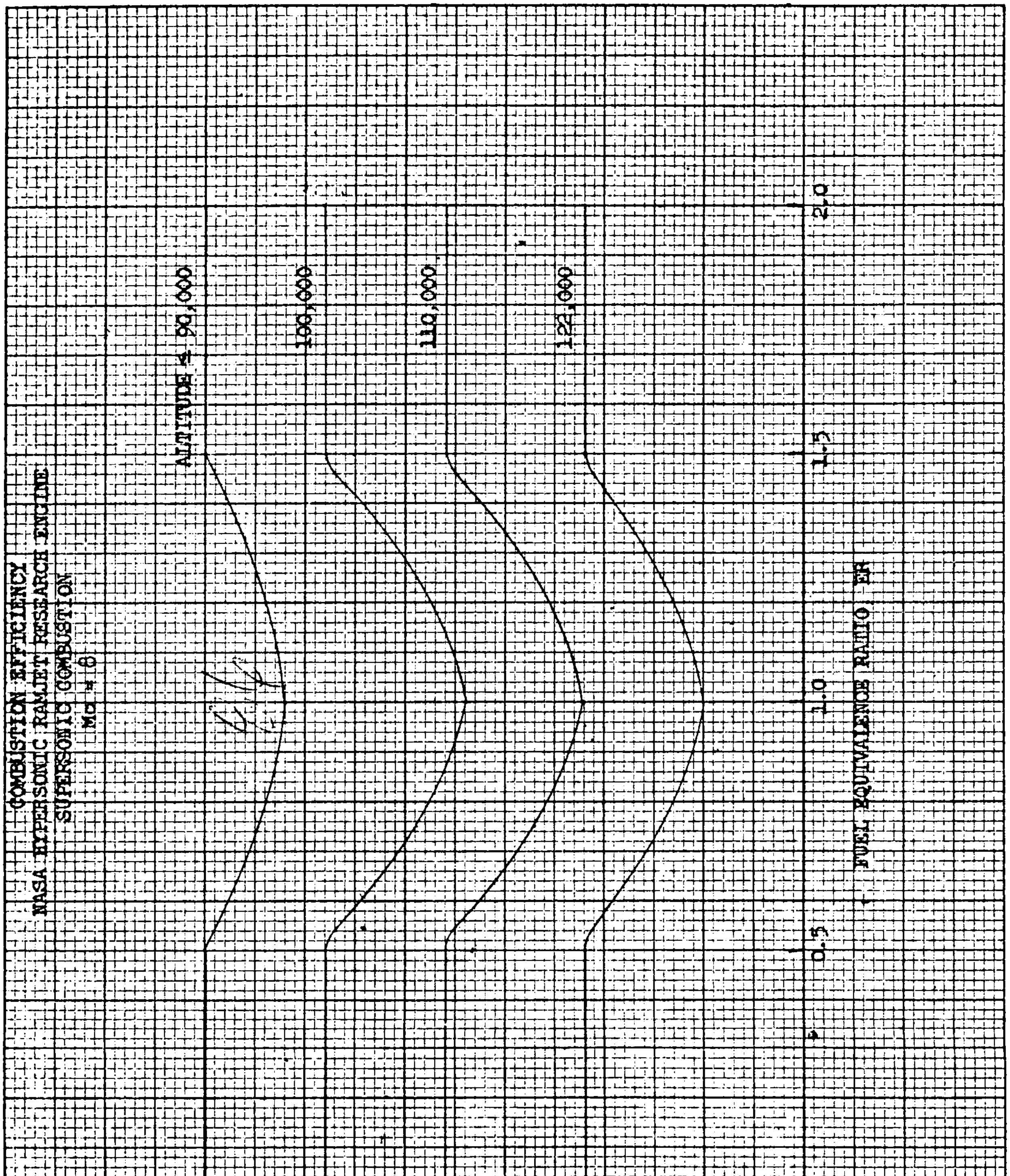
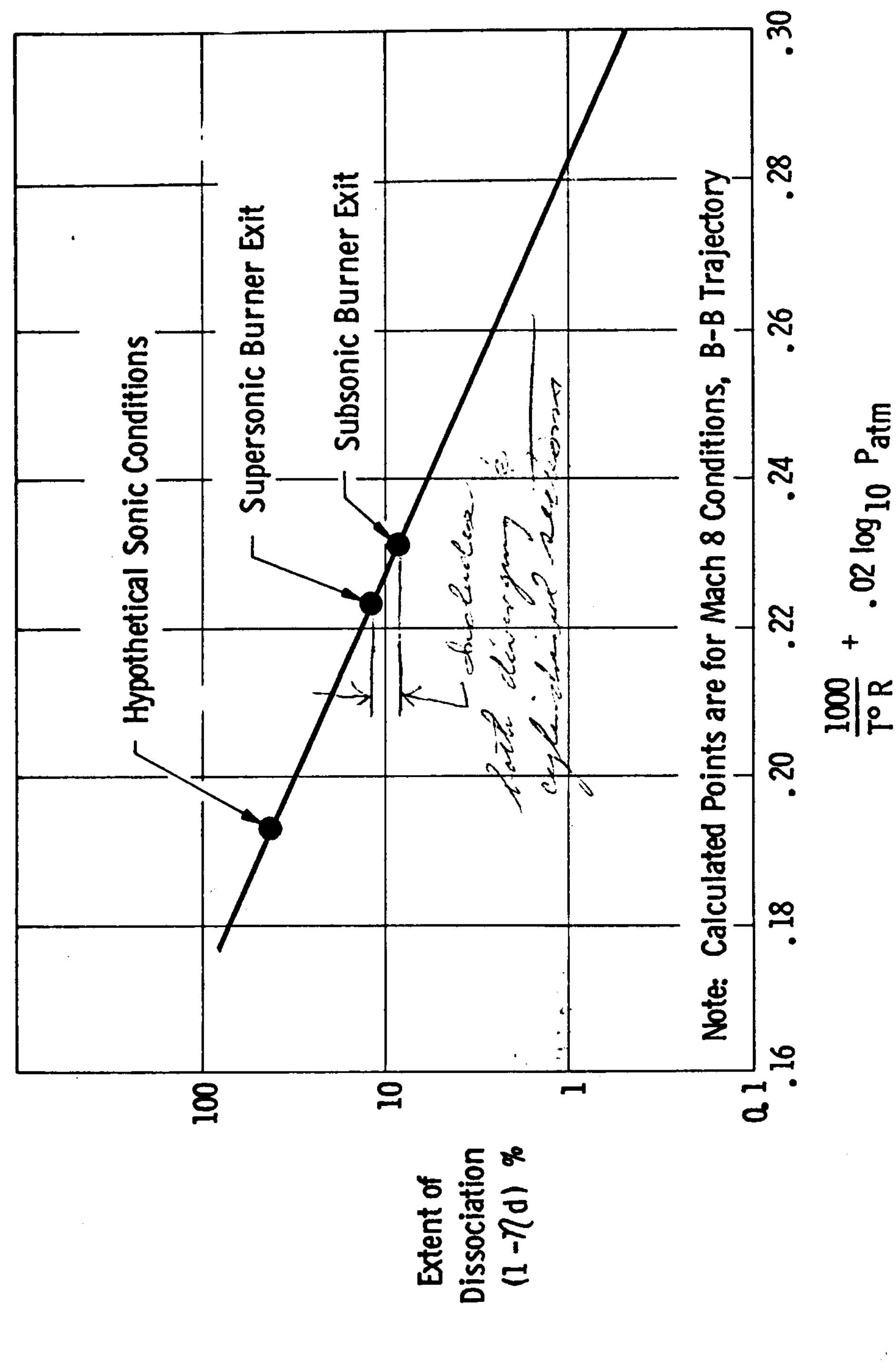

COMBUSTION EFFICIENCY
NASA HYPERSONIC RAMJET RESEARCH ENGINE
SUPERSONIC COMBUSTION
Mo = 8
ALTITUDE ≤ 90,000
100,000
110,000
122,000
0.90
0.80
0.70
0.60
0
0.5
1.0
1.5
2.0
FUEL EQUIVALENCE RATIO, ER
COMBUSTION EFFICIENCY, ηB

Figure 65



Equilibrium Dissociation Level

Figure 66

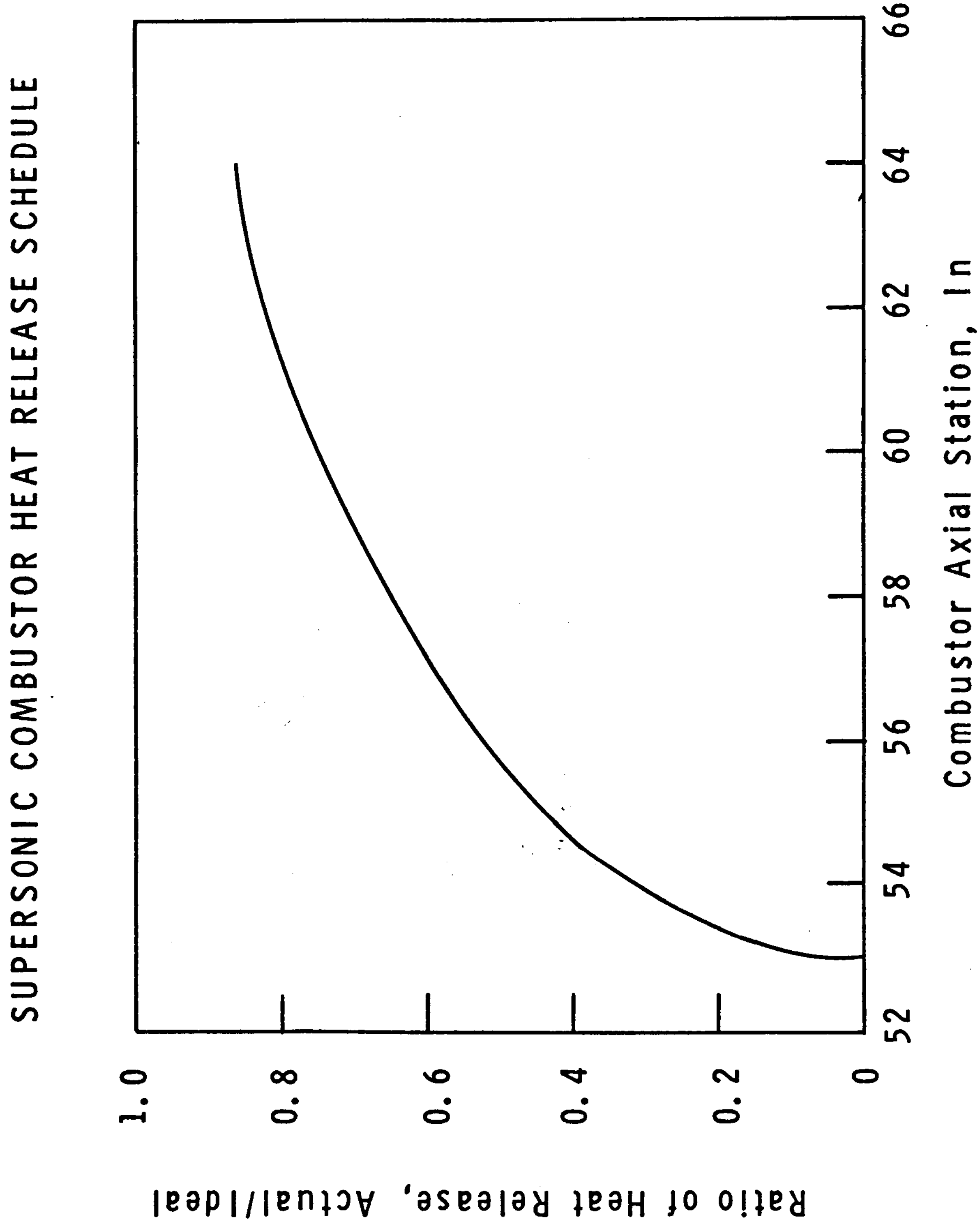

SUPERSONIC COMBUSTOR HEAT RELEASE SCHEDULE
Ratio of Heat Release, Actual/Ideal
1.0
0.8
0.6
0.4
0.2
0
52
54
56
58
60
62
64
66
Combustor Axial Station, In

Figure 67:

INDUCTION LENGTH VERSUS FLIGHT MACH NUMBER

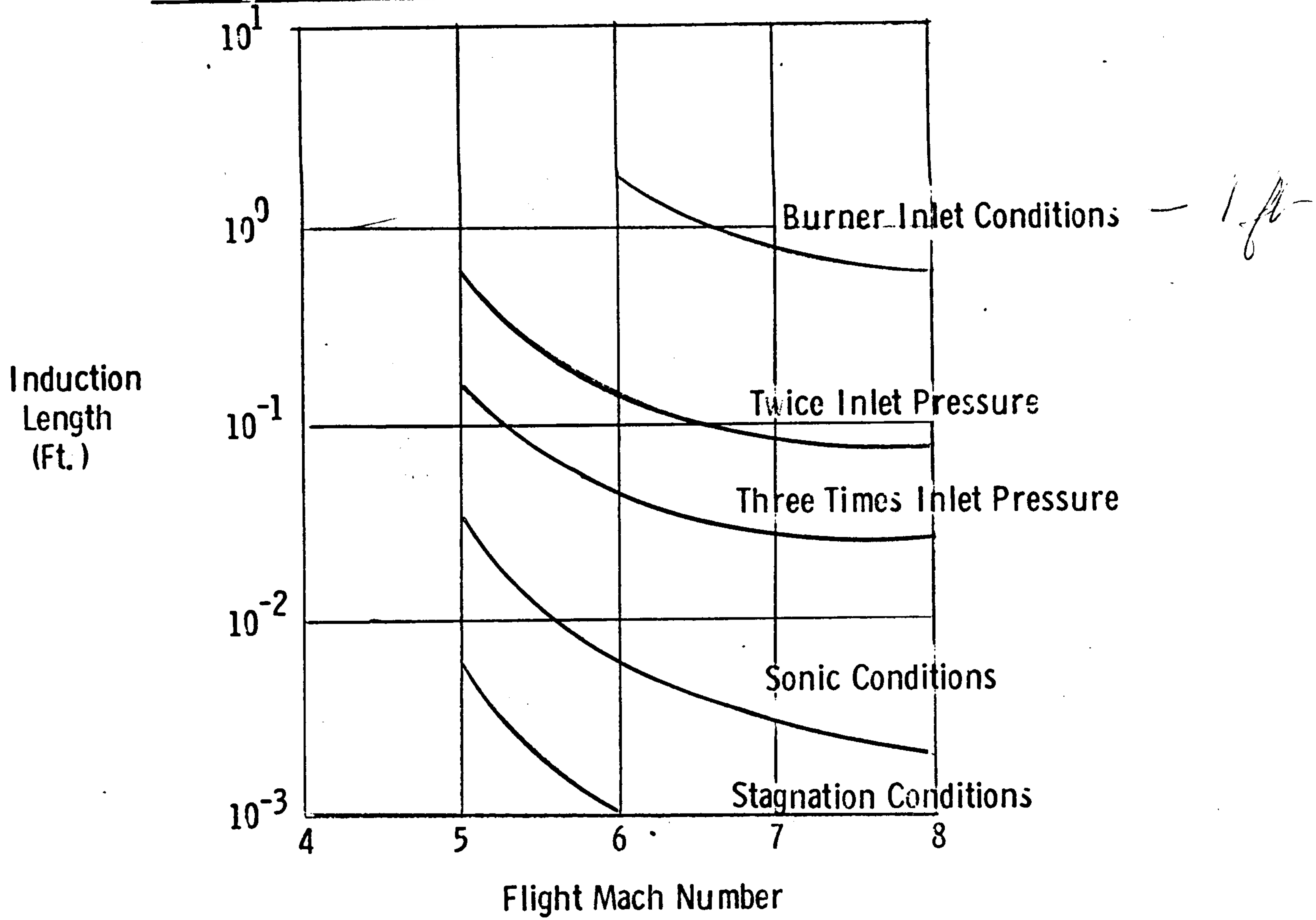


Figure 68

SHOCK WAVE IN SUBSONIC DIFFUSER

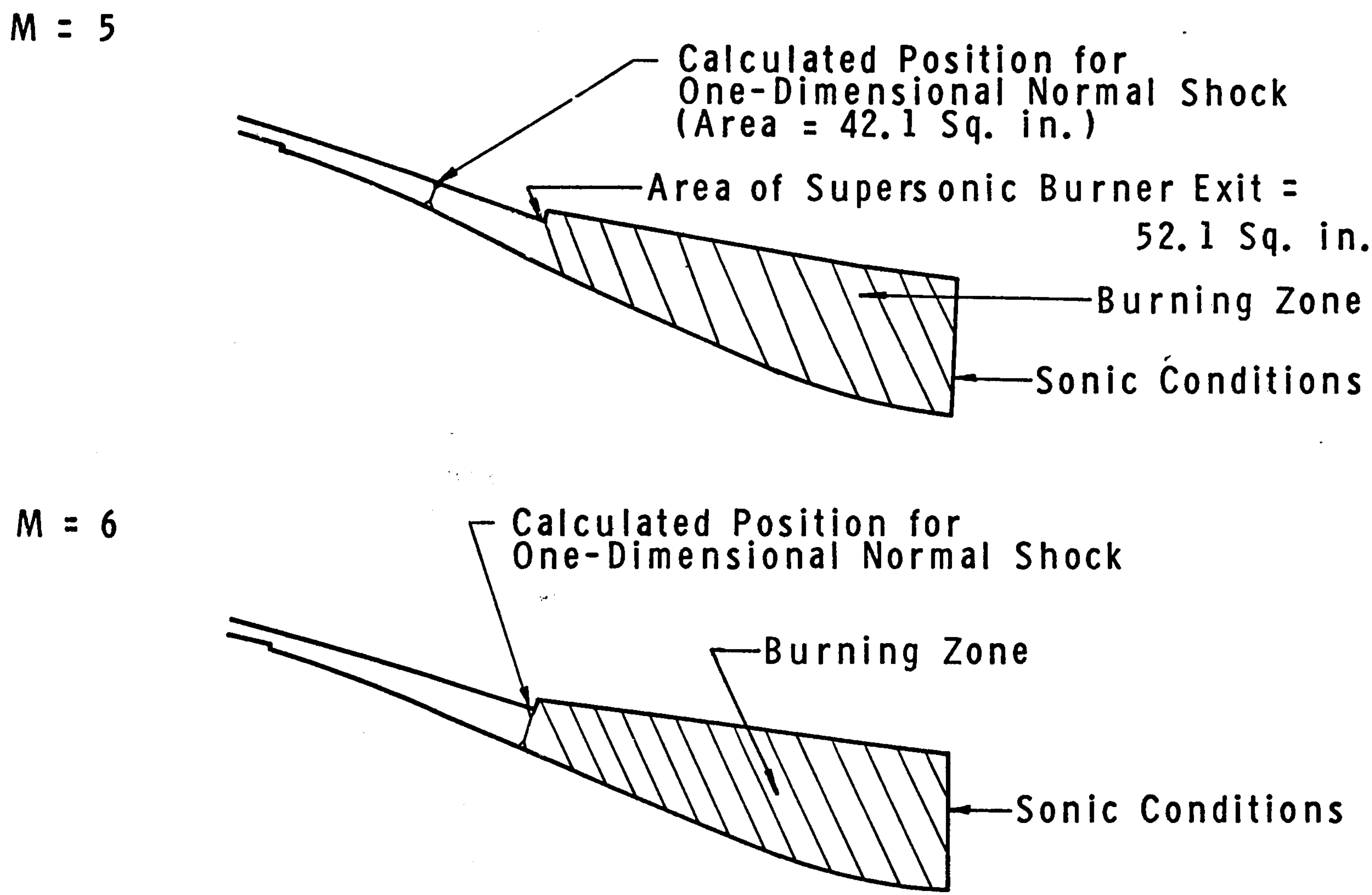
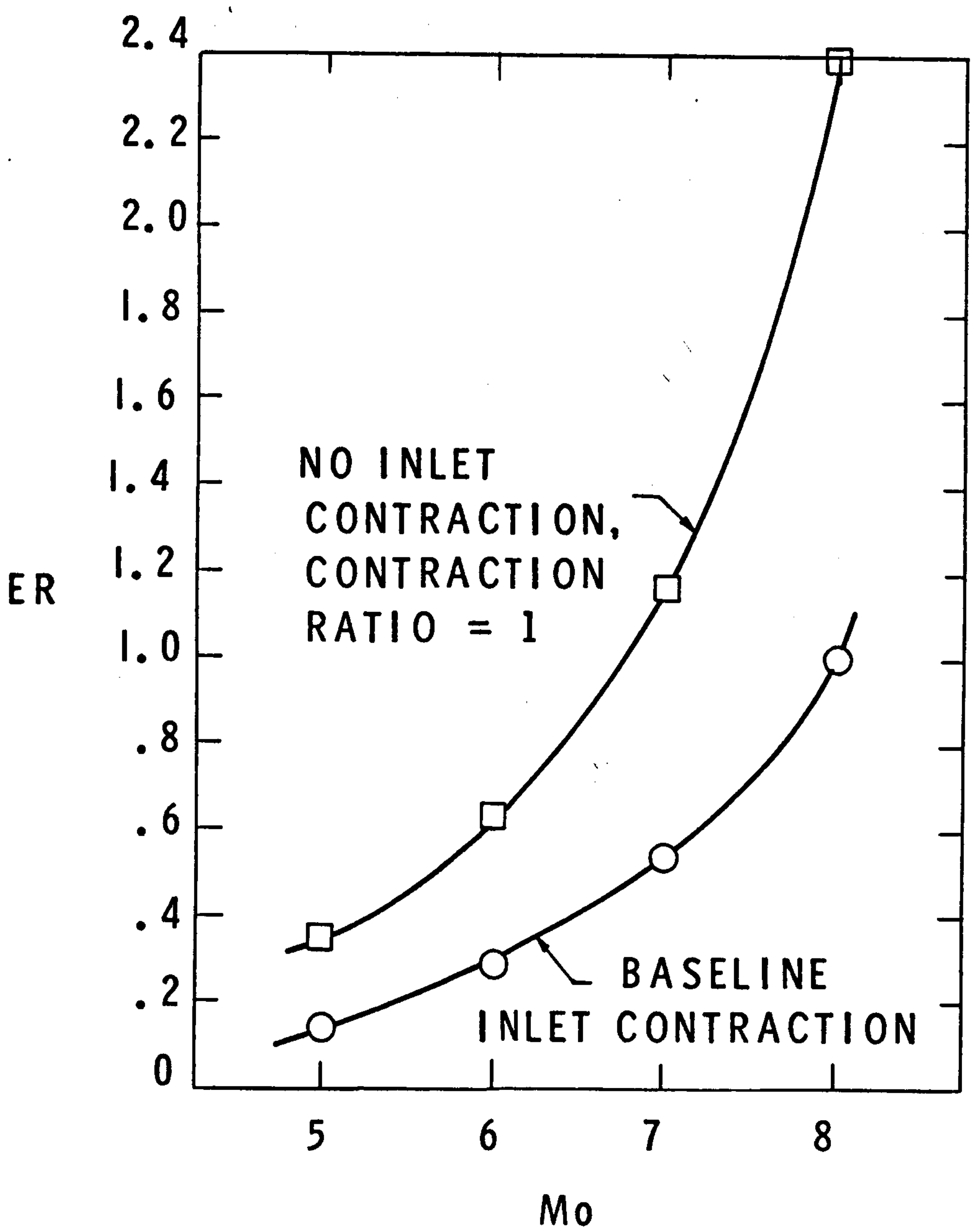


Figure 69

CHOKING EQUIVALENCE RATIO IN CONSTANT-AREA BURNER



- NORMAL INJECTION
- COOLING REQUIREMENT FUEL TEMPERATURE
- REAL GAS
- FRICTIONLESS

Figure 70

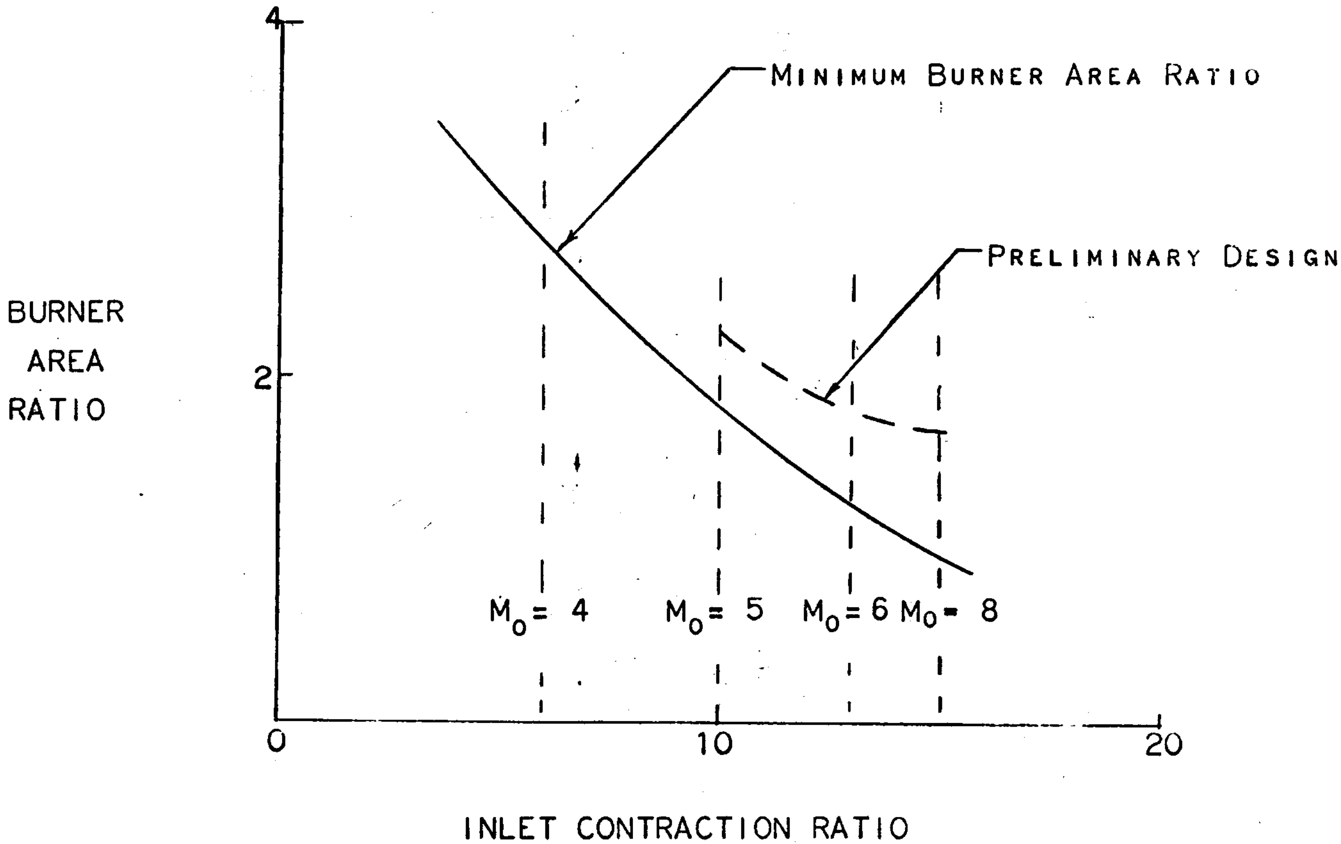


FIGURE 71: BURNER AREA RATIO IN EARLY PORTION OF BURNER

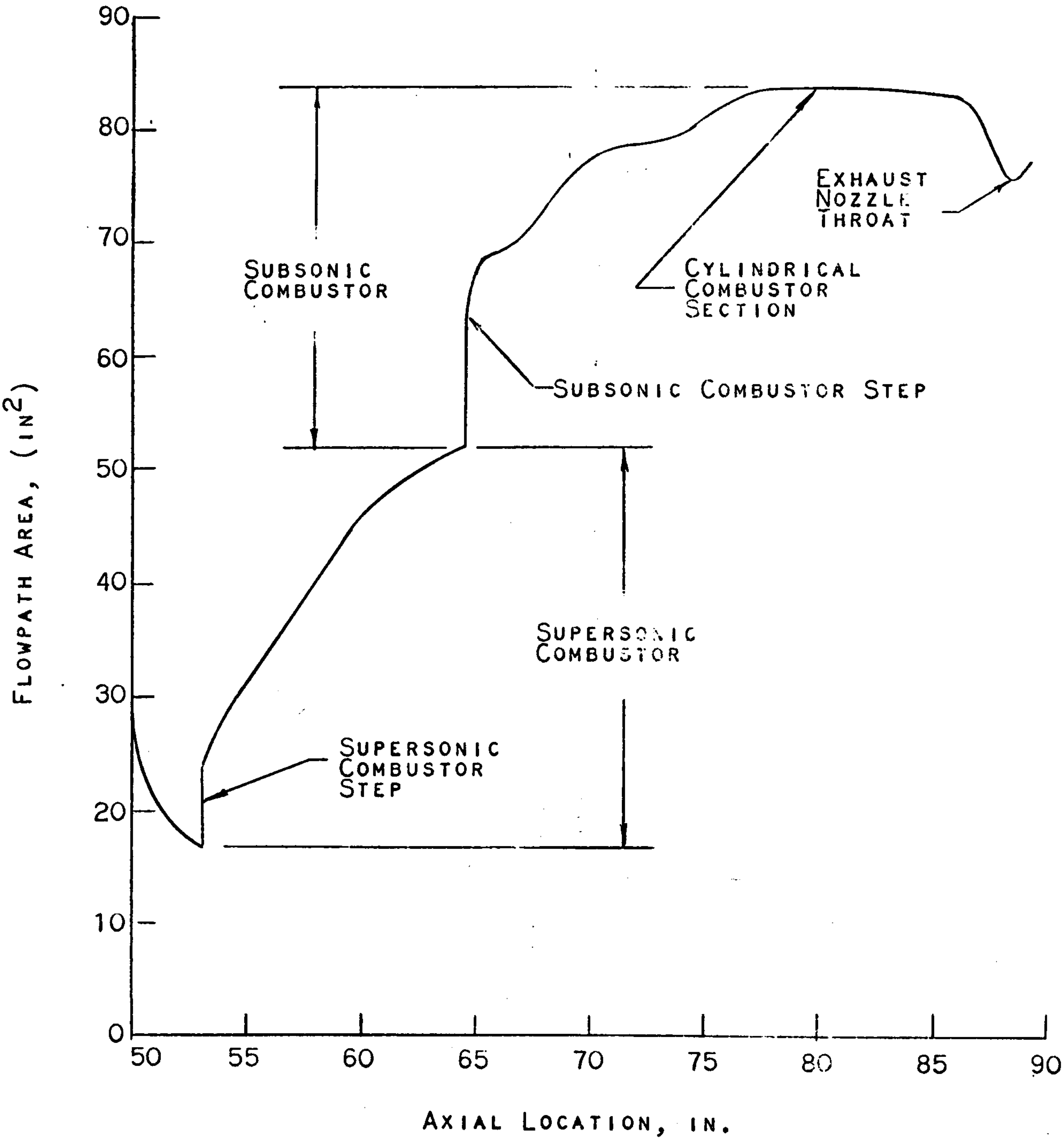


Figure 72: Mach 8 Flow Areas Used in Cycle Calculations

INLET THROAT STEP PRESSURE RISE

$M_0 = 8$

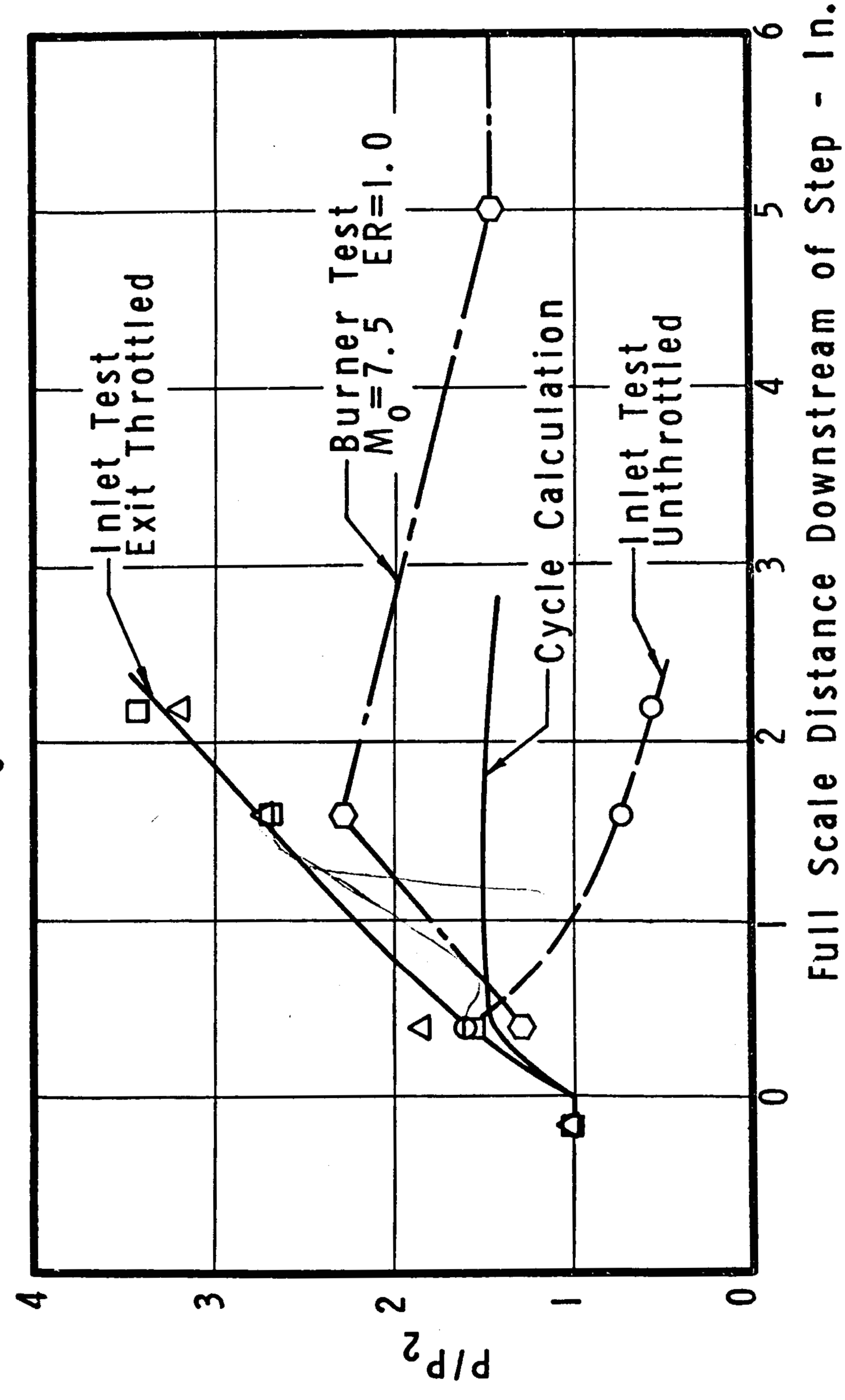


Figure 73